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Eurospace

We have waited so long for a Government decision on a British Space Programme that we have almost—but not quite—reached the point where we would welcome *any* decision, even an unfavourable one. Indeed, a cynic might suggest that that provides a possible explanation for the prolonged delay—it is intended to ensure that the eventual announcement of our final abandonment of any interest in space technology will be met with resigned relief rather than with howls of protest.

We make no such suggestion, nor (although we are acquainted with the story of the potatoes in the cookhouse) do we support the alternative hypothesis that those in authority delay making a decision because they are incapable of making a decision.

Instead we incline towards the view that for some time now it has been recognized that British participation in astronautics is unavoidable, no matter how unpalatable that may be to some members of the Government and certain of their advisers. We ourselves would certainly consider such participation inevitable if Britain is to continue along the main stream of history and attempt to remain a leading industrial nation. But then we are both prejudiced and better acquainted with our subject.

If it has at last been realized that Britain must enter space, why are we making haste so slowly? Presumably the reluctance is attributable to a desire to avoid too great a commitment, either financial or in man-power. It is hoped that we can persuade others to share the burden with us.

That is all very well up to a point. For some years the Society has consistently advocated that Britain should join with the Commonwealth and Europe in undertaking a co-operative space programme. We therefore welcome the agreement signed in Meyrin, Switzerland, at the beginning of this month, setting up a preparatory commission to study the possibility of creating a permanent European organization for collaboration in space research. This agreement was the outcome of intergovernmental discussions carried out in G.E.E.R.S. (Groupe d'Études Européen de Recherches Spatiales; European Study Group for Space Research).

While welcoming the agreement as necessary, we do not consider that it is sufficient. How could we, when the proposed initial budget is only 935,000 N.F., or about £68,000 (of which the United Kingdom is to provide 25%)? For an intergovernmental body, that will hardly cover the cost of offices, filing cabinets and stationery, let alone a worthwhile space programme.

The reason for such a small budget is not hard to find; it appears to cover only the cost of discussing whether to have a co-operative space programme or not. If it is decided to proceed further, no doubt the expenditure will be much greater, although still not as great as we should consider satisfactory.

This is because although the preamble to the agreement mentions "collaboration in research in space science and space technology," it is not unlikely that this body will restrict at least its initial activities to space science. Thus, the United Kingdom delegate, Mr. R. N. Quirk, said that the U.K. could not subscribe to the idea of creating an organization that would develop large rockets. France and Sweden also opposed this idea, although the Italian, Dutch, Belgian and Swedish delegates were in favour of widening the terms of reference to include launcher development.

Unable to penetrate the thick fog surrounding the Government's deliberations on space, we can only guess why there should be opposition to European co-operative rocket development, and our guess is that it is thought undesirable that more countries should become better acquainted with rocket technology—this for commercial rather than military reasons. The U.K. government might well be expected to prefer selling Blue Streak to the Eurospace organization rather than scrap the British rocket, and then have to pay part of the development cost of a co-operative vehicle.

Our interpretation is probably wrong—we hope that it is, for such an approach would surely be doomed to failure. In our opinion there *should* be a co-operative European vehicle programme, and the decision of the commission in charge of the programme as to whether to support the continued development of Blue Streak or not should be made purely on a consideration of the vehicle's own merits—and we are convinced that it has many. In the meantime, it is the Government's duty to go ahead on its own with the proposed Blue Streak/Black Knight launcher and thus demonstrate Britain's confidence in its own product. We should also be engaged in the design of a communications satellite.

However, we are one with Mr. Quirk as regards G.E.E.R.S. and its successors. G.E.E.R.S. is a scientific, rather than technological body, and can well restrict its activities to space physics. What is needed is another body to organize Eurospace activity in the technological and commercial fields—perhaps this body might well be an agency of O.E.E.C. Such a body could then take over the responsibility for Spadeadam and similar establishments, and allot contracts to industry for specific parts of the programme.

Such contracts should not, of course, be confined to firms of only a few of the countries participating in the Eurospace project—we want every country to participate in the project industrially as well as financially. The range of activities which would have to be undertaken in the type of programme we have in mind (say, the development of a wholly European communications satellite and launcher) is so wide that each of the participating countries could find at least one task to tackle. Not only the aircraft and electronics industries, but also metallurgical, chemical, instrument, plastics, and civil engineering firms have a part to play. We hope to hear from these at the forthcoming European Spaceflight Symposium.

G. V. E. THOMPSON.

STRESSES ASSOCIATED WITH LUNAR LANDINGS*

By Professor JOHN S. RINEHART,† Ph.D.

(Communication from the Colorado School of Mines)

ABSTRACT

The probable stresses which would be developed during lunar impact landings are discussed phenomenologically and quantitatively. For landings against rocks in the velocity range from a few hundred feet per second to a few thousand feet per second, the stress is proportional to the first power of the impact velocity, with its magnitude depending upon the particular rock struck and increasing approximately linearly with shear strength and with specific acoustic resistance, but ranging roughly from 6000 lb./in.² at 100 ft./sec. to 2,000,000 lb./in.² at 6000 ft./sec. In landings against loose soils the stress, much lower than for rock impacts, is substantially independent of velocity at low velocities, but begins to depend strongly on the square of the velocity as the velocity is increased. The excursions of a 10,000-lb. vehicle into rock and soil surfaces are compared.

I. INTRODUCTION

PROPOSED astronautical ventures envision impact landings against rock effected at moderate and high velocities. It is imperative in designing suitable vehicles to have an appreciation for the nature and magnitudes of the impulsive transient stresses which will be associated with such landings. This paper discusses these stresses, the discussion being based largely upon results obtained from experimental investigations of impact against rock recently carried out at the Mining Research Laboratory, Department of Mining Engineering, Colorado School of Mines.

II. FORCES OPERATIVE DURING IMPACT

When an object such as a lunar landing vehicle, missile or other projectile, moving at considerable velocity, strikes a massive target such as the Moon's surface, intense complex transient stress situations develop within both the vehicle and the object against which it is striking. Usually the struck body fails, the missile or projectile penetrating into the target to some depth, where it comes to rest or is forcibly ejected from its burrow by expansion of a plug of target material compressed in front of it. When the impact velocity is very high, the projectile itself may fail, breaking apart or becoming distorted. The nature of the stresses that are set up in the vehicle while it is moving through the target before it comes to rest depends largely upon how the target fails, the decelerating force in some instances being constant during the entire excursion of the projectile and in others, depending in a simple or complex way upon the instantaneous value of the velocity.

Many experimental studies[1–3] have been carried out to determine the nature of these forces, some of the earliest being the French Army experiments performed at Metz between 1835 and 1845.[4] The strategem in most instances has been to measure depth of penetration and deduce force relationships from this observation rather than to perform the more difficult, usually almost impossible, feat of measuring stresses during penetration.

In many materials, the force is found to be the sum of two components: a constant force, independent of the velocity, representing some inherent strength of the material; and a component, proportional to the square of the velocity, representing inertial forces. For these materials, the average force per unit area acting on the projectile at any instant may thus be written

$$\frac{F}{A} = a + bv^2 \quad .. \quad .. \quad (1)$$

where v is the velocity of the projectile at that instant, A, the cross-sectional area of the penetrating projectile taken normal to its trajectory; and a and b are constants, dependent upon the target material and the shape of the projectile. It follows that the total penetration s is given by

$$s = \frac{m}{2bA}\log_e\left(1 + \frac{b}{a}v_0^2\right) \quad .. \quad .. \quad (2)$$

where v_0 is the striking velocity of the projectile, and m its mass.

Values of a and b for spherical projectiles impacting in a loose sand-gravel mixture and compacted earth were obtained in the Metz experiments. For sand-gravel, a and b are respectively, 620 lb./in.² and 0·0115 (lb./in.²)/(ft./sec.)²; and for compacted earthworks, 432 lb./in.² and 0·0008 (lb./in.²)/(ft./sec.)², respectively. Fig. 1 shows the average stresses which would be acting on spherical projectiles penetrating these two materials as a function of impact velocity, broken down into its two components: velocity-independent and velocity-dependent. Note that for sand-gravel, in going from 100 ft./sec. impact velocity to 1000 ft./sec. the dominant force term changes from a static term of 620 lb./in.² to a dynamic inertial term of 11,500 lb./in.² In earthwork the transition takes place at a much higher velocity, illustrating that in hard compact materials the inertial

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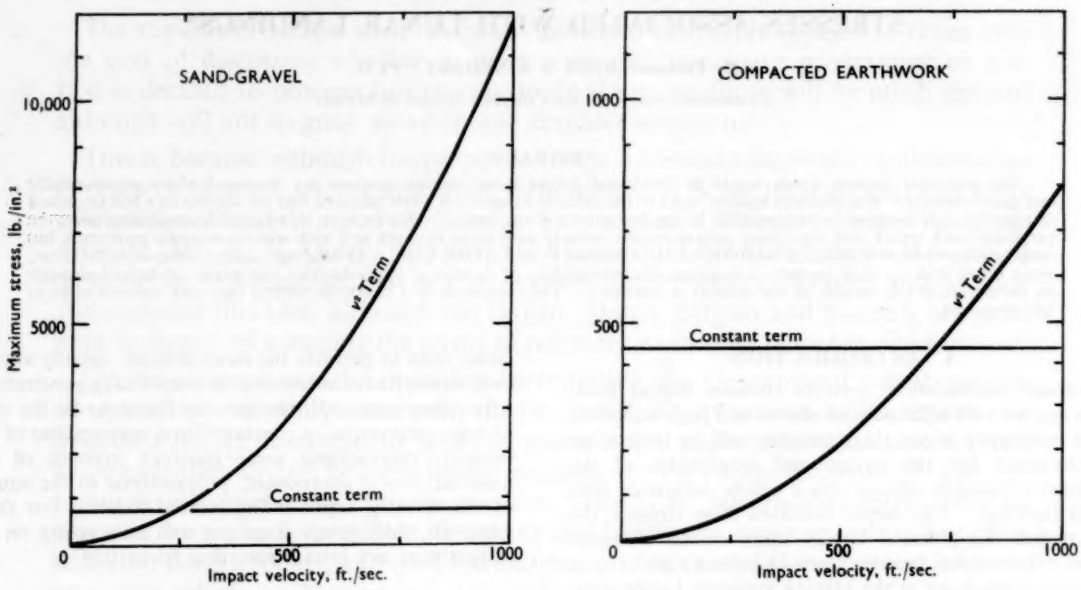

FIG. 1. Relative values of velocity-dependent and velocity-independent components of force resisting penetration.

velocity-dependent term does not become relatively significant until high velocities are reached. In steel, for example, it is negligible up to velocities of about 3000 ft./sec. At very high velocities (thousands of feet per second) it is of paramount importance.

Experiments on rocks show clearly that the dominant force resisting penetration into those materials is neither constant nor a velocity-squared term, the principal force component being proportional to the first power of the velocity. In this case the total penetration can be written

$$s = K_1 \frac{m}{A}(v_0 - d) \quad .. \quad .. \quad (3)$$

where K_1 is a constant dependent upon target material and the shape of the projectile and d is a small, frequently negligible, correction factor, very complex in nature. During penetration the instantaneous value of the force on the penetrating projectile will be proportional to the velocity at that instant.

A force proportional to the first power of the velocity could arise in two ways: as an elastic compression or as a viscous or shearing drag. During elastic compression, resulting from impact, the stress σ is given by

$$\sigma = \rho c v \quad .. \quad .. \quad .. \quad (4)$$

where ρ is the density of the target material; c, the velocity of propagation of a longitudinal wave in it; and v, the velocity of compression. The exact nature of the viscous or shearing drag is somewhat more difficult to understand. In the experiments to be discussed later it was found that among several rocks the force resisting penetration correlates well with both shearing strength and the specific acoustic resistance, the product ρc, of the rock. Progress through the material might logically proceed as a series of repeatable events in which the material is first compressed, more or less elastically, generating a stress proportional to velocity, which shears the material, exposing a new region for the process to repeat itself at a somewhat reduced stress level.

III. LATERAL STRESS

A landing vehicle would be, until it is brought to rest, the centre of a dynamically changing force field. The effect of this force is felt both vertically and laterally. As far as lateral effects are concerned, the moving force generates, in addition to its vertical component, lateral stresses consisting of a series of cylindrically expanding transient stress waves, in many circumstances capable of damaging the material. The main effect of these lateral stresses will be to disrupt the target material. If the target material deforms easily, it will flow radially, forming a crater; if it is brittle, a crushed region will appear. Radial cracks will usually develop as a consequence of the hoop stresses set up at the front of the expanding stress wave. For many reasons it is difficult to describe the stress situation precisely and in detail but fortunately this is not necessary in order to describe the principal stresses developed within the impacting body.

IV. PENETRATION IN ROCKS

Recent tests at the Mining Research Laboratory, Department of Mining Engineering, Colorado School of Mines, in which spherical steel projectiles were fired into rocks, have not only thrown considerable light on

the mechanics of crater formation in rock but have also yielded much quantitative data on the nature and magnitudes of the impact forces developed.

Specifically, tests were run using $\frac{3}{16}$-in. diameter, 0·442-g., and $\frac{9}{32}$-in. diameter, 1·503-g. steel balls fired from 0·22-in. and 0·30-in. calibre guns, respectively, at velocities ranging from 300 to 6,000 ft./sec. The rock targets were a poorly cemented, argillaceous, feldspathic sandstone, ranging from thinly bedded to massive structure; a holocrystalline, coarse-grained inequigranular, quartz-biotite granite; Carlsbad potash; and tuff.

The typical crater produced in sandstone, potash, and tuff, by a projectile fired at an impact angle of 90° is shown in Fig. 2. As the projectile penetrates the rock it crushes the material in front of it, forcing the crushed material aside, and forming a long cylindrical hole, or burrow, of the same diameter as the projectile, surrounded by a zone of compressed crushed target material. The upper part of the crater, formed as the material is broken and the chips and debris blown from the face of the rock, has a cup shape.

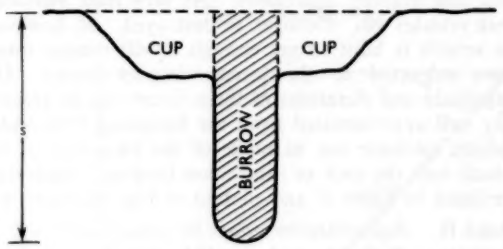

FIG. 2. A typical impact crater in rock.

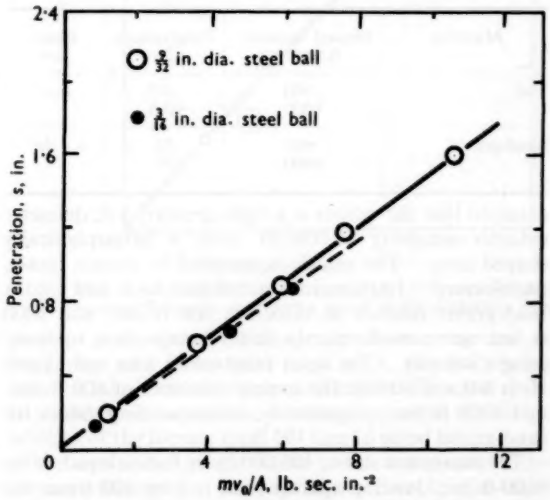

FIG. 3. Penetration *versus* impact velocity curve for sandstone. Projectiles of two sizes were used to obtain data, the abscissa being brought to the same scale by multiplication by the factor (m/A), m being the mass of the projectile and A its cross-sectional area.

The penetration s was found to vary linearly with impact velocity v, as can be seen in Fig. 3, 4, and 5, the exact relationship being given by Equation (3). The applicability of this relationship implies at once that inertial forces are negligible, the instantaneous force F resisting the projectile being given by

$$F = \left(\frac{A}{K_1}\right) v \qquad \ldots \quad \ldots \quad (5)$$

and the instantaneous stress σ, averaged over the front surface of the projectile, by

$$\sigma = \frac{v}{K_1} \qquad \ldots \quad \ldots \quad \ldots \quad (6)$$

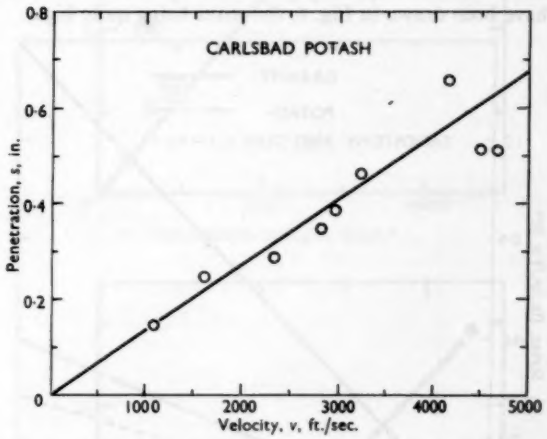

FIG. 4. Penetration *versus* impact velocity curve for Carlsbad potash. Projectile was $\frac{3}{16}$-in. diameter steel ball.

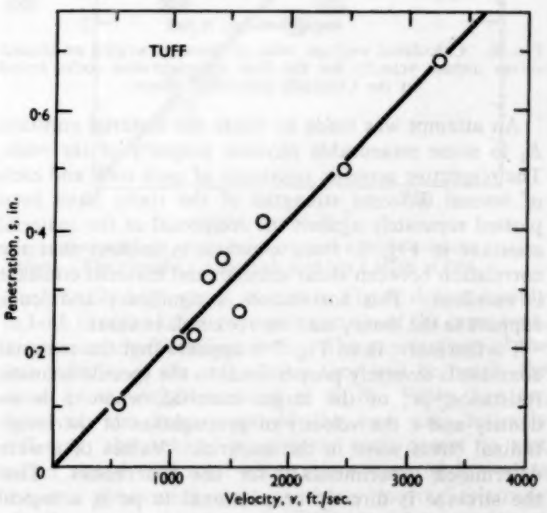

FIG. 5. Penetration *versus* impact velocity curve for Oak Springs tuff. Projectile was $\frac{3}{16}$-in. diameter steel ball.

TABLE I.—*Values of K_1 and stress developed per foot per second of impact velocity*

Rock	K_1 (ft./sec)/(lb./in.²)	$1/K_1$† (lb./in.²)/(ft./sec.)
Carlsbad potash ..	0·010	100
Granite	0·004	250
Sandstone ..	0·019	53
Oak Springs Tuff	0·016	56

† Stress per foot per second of impact velocity ($=1/K_1$).

Values for K_1 for the four rocks are listed in Table I. From these values it is possible to calculate the maximum stress which an impacting sphere will experience. Curves have been drawn in Fig. 6, the stress being quite high.

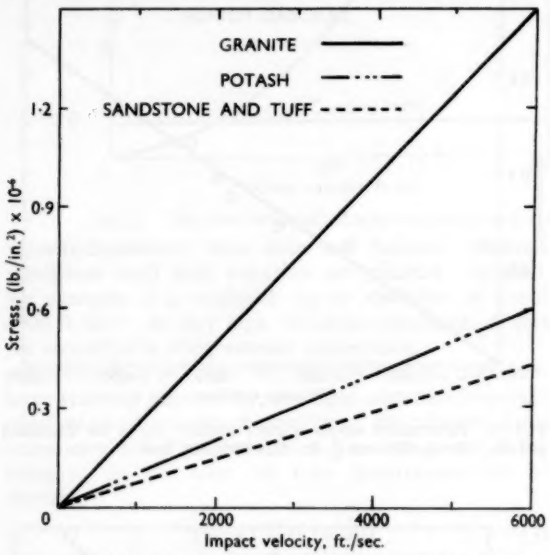

FIG. 6. Calculated average value of stress generated on impact *versus* impact velocity for the four representative rocks tested at the Colarado School of Mines.

An attempt was made to relate the material constant K_1 to some measurable physical property of the rock. The respective acoustic resistance of each rock and each of several different strengths of the rocks have been plotted separately against the reciprocal of the material constant in Fig. 7, from which it is evident that the correlation between shear strength and material constant is excellent. This correlation is significant and lends support to the theory that the rock fails in shear.

Furthermore, from Fig. 7 it appears that the material constant is inversely proportional to the specific acoustic resistance, ρc, of the target material, where ρ is its density and c the velocity of propagation of the longitudinal stress wave in the material. Values of c were determined experimentally for the four rocks. That the stress σ is directly proportional to ρc is a logical result, since in elastic impact the stress set up in the impactor and the target depends upon the relative acoustic resistances of the two. For a theoretically completely rigid impactor having velocity v and striking a target possessing acoustic resistance, ρc, the stress σ set up in the target is

$$\sigma = \rho c v$$

The projectile here is spherical so that a shape factor having a value of about 0·3 enters, the average particle velocity in front of the projectile being less than the velocity of the projectile. For a blunter projectile, the value of the shape factor should increase, reaching almost one for a right circular cylinder.

Experimentally, it was found that the projectile usually did not remain in the crater but bounced back out with a velocity of a few feet per second. It is easy to imagine that ejection of the ball follows expansion of a compressed plug of rock, lying just ahead of the projectile when it comes to rest.

V. APPLICATION OF RESULTS TO LUNAR LANDINGS

Moon-bound vehicles may land on rock or on loose sand and gravelly aggregates. At very high velocities weak vehicles will, of course, be destroyed. If, however, the vehicle is built strong enough it will remain intact when subjected to the intense impact forces. The magnitude and durations of these forces can be reasonably well approximated from the foregoing data and a realistic estimate can be made of the excursion of the vehicle into the rock or sand upon landing. Such data are listed in Table II and plotted in Fig. 8, where it is assumed that the vehicle is a right circular, 2-ft. diameter cylinder weighing 10,000 lb. with a hemispherically shaped nose. The vehicle is assumed to remain intact, undeformed. Impacts into a tuff-like rock and into a sand-gravel mixture at velocities 600 ft./sec. and 6000 ft./sec. are considered, the latter being close to lunar escape velocity. The total penetration into tuff (Table II) is 6·0 and 60·0 ft. for impact velocities of 600 ft./sec. and 6000 ft./sec., respectively, corresponding values for sand-gravel being 62 and 195 ft.

TABLE II.—*Representative values for penetration for 2-ft. diameter cylindrical nondestructible vehicle with hemispherically shaped nose, weighing 10,000 lb.*

Material	Impact velocity, ft./sec.	Penetration, ft.	Force law
Tuff	600 6000	6·0 60·0	$K_1 v$
Sand-gravel ..	600 6000	62 195	$a + bv^2$

The maximum stress, 400,000 lb./in.², developed in the 6000-ft./sec. landing against rock is over 100 times the stress of 3000 lb./in.² developed in the corresponding landing against sand. The maximum deceleration of the vehicle is about 10,000 *g* for rock impacts and 100 *g* for sand. The deceleration of the vehicle will be

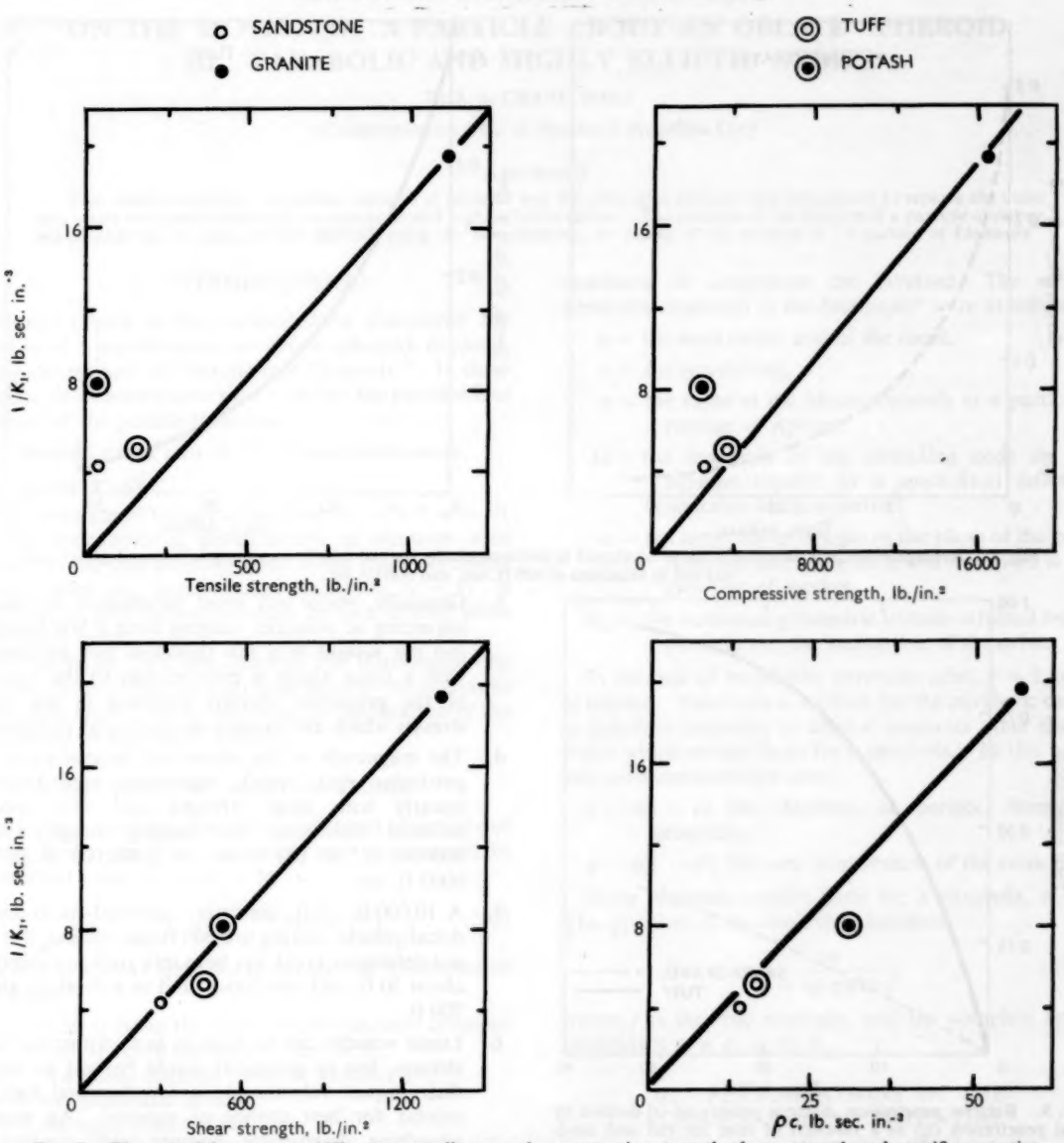

FIG. 7. The material parameter $1/K_1$ *versus* tensile strength, compressive strength, shear strength and specific acoustic resistance for the four rocks tested at the Colorado School of Mines. The straight line in each case has a slope of one.

directly proportional to the ratio of its projected or impacting area to its total mass, m/A, the ratio being least for long slender vehicles and highest for squat ones. It would seem advisable, whenever possible, to design vehicles of the former type.

The stresses are proportional to the first power of the impact velocity for landings against rock; for sand impacts, the stresses involve both a constant term independent of velocity, important at low velocities, and a velocity-squared term, important at high velocities. The behaviour of the vehicle is instructive to follow. The initially high stress, plotted in Fig. 8, decays rapidly in both cases to a fairly low value in about the same time, approximately 30 milliseconds. The penetrations of the two vehicles at the end of this time will be nearly the same, in the neighbourhood of 50 ft. From then on, however, the behaviours of the two are distinctly different: the vehicle penetrating rock comes to rest rather shortly, its total penetration being 60 ft.; while the one penetrating sand continues moving relatively slowly, not coming to rest until it has burrowed in a distance of about 200 ft. (Fig. 9).

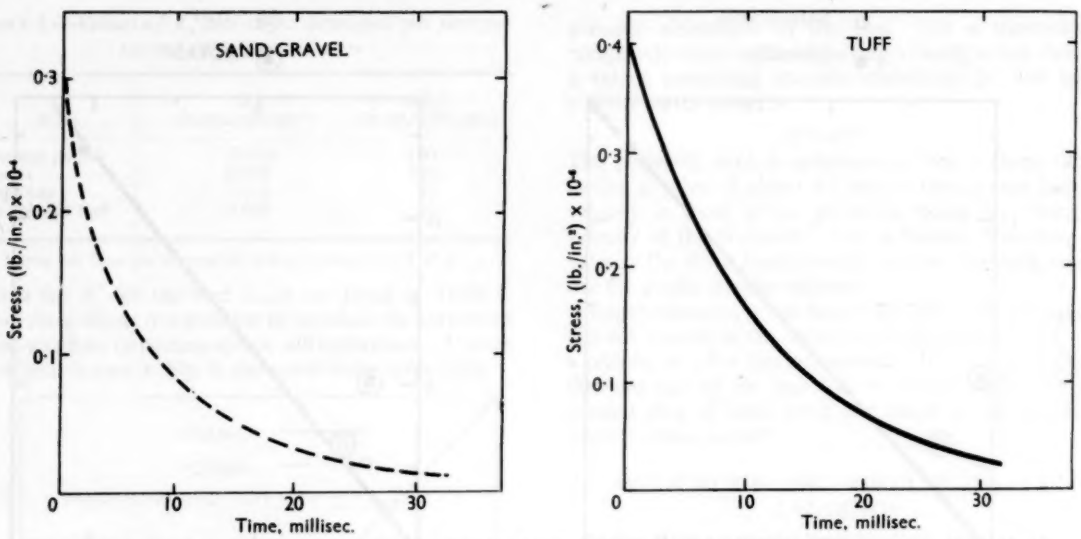

FIG. 8. Average stress as a function of time developed in hemispherically shaped missile impacting in (*a*) sand-gravel and (*b*) tuff at velocities of 600 ft./sec. and 6000 ft./sec.

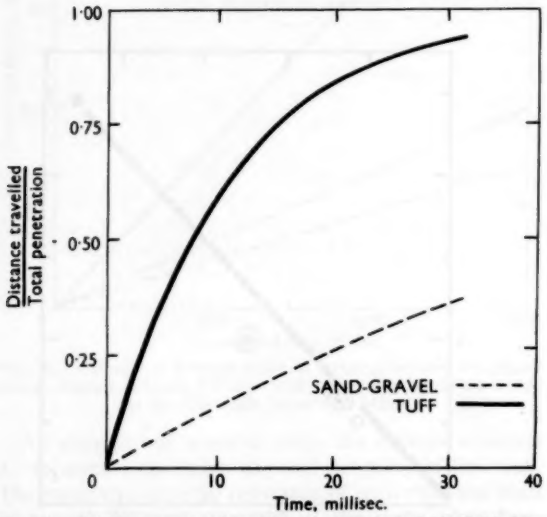

FIG. 9. Relative penetration, distance penetrated (s) divided by total penetration (s_0) as a function of time for tuff and sand-gravel.

VI. SUMMARY

1. The impact at moderate velocities of a vehicle against rock generates, in general, a highly complex transient stress situation which can lead to failure of both the vehicle and the rock against which it impacts.
2. Under impact loading, various materials behave quite differently depending upon whether they are coherent or non-coherent, brittle or plastic, and upon their relative strengths.
3. Generally, rocks will resist penetration by bodies impacting at velocities ranging from a few hundred feet per second to a few thousand feet per second with a force which is proportional to the velocity of the projectile, thereby inducing in the body stresses which are likewise proportional to velocity.
4. The magnitude of the stress will depend upon the particular rock struck, increasing approximately linearly with shear strength and with specific acoustic resistance, but ranging roughly from 6000 lb./in.² at 100 ft./sec. to 2,000,000 lb./in.² at 6000 ft./sec.
5. A 10,000-lb., 2-ft. diameter, spherical-nosed, cylindrical vehicle striking at 6000 ft./sec. would, if it did not deform or break up, bore into rock to a depth of about 50 ft. and into loose sand to a depth of about 200 ft.
6. Lunar vehicles can be built so as to withstand these stresses, but in general it would be best to reduce their impact velocities to a few hundred feet per second for best chance of survival. An energy-absorbing, destructible, bumper could significantly reduce the stresses.

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ON THE MOTION OF A PARTICLE ABOUT AN OBLATE SPHEROID
III. PARABOLIC AND HIGHLY ELLIPTIC ORBITS*

By J. A. CRANE, B.Sc.†

(Communication from de Havilland Propellers Ltd.)

ABSTRACT

Two orbital elements—the radial distance at perigree and the semi latus rectum—are introduced to replace the usual semi major axis and eccentricity for parabolic and highly elliptic orbits. The solution of the motion of a particle about an oblate spheroid, *in vacuo*, is then derived, using the new elements, by means of the method of "Variation of Elements".

I. INTRODUCTION

PREVIOUS papers in this series[1,2] have considered the motion of a particle about an oblate spheroid, *in vacuo*, using the method of "Variation of Elements." In these papers, six elements were used to define the position and velocity of the particle including:

a the semi major axis of the instantaneous conic;

e the eccentricity.

The presence of factors in the form $(1-e^2)^{-m}$, $m > 0$, in the derivatives of perturbations in elements with respect to the true anomaly leads to the theory breaking down when an instantaneous parabola ($e = 1$) is encountered and the computations become difficult when e is nearly unity.

In this paper, two new elements are introduced—the perigee distance and the semi latus rectum—to overcome these difficulties, thereby widening the application of the "Variation of Elements" method.

II. THE OSCULATING CONIC

It is again assumed that the gravitational potential due to the Earth—an oblate spheroid—can be expressed to a sufficient order of accuracy by:

$$\Psi = \frac{\mu}{\delta}\left\{\left(\frac{\delta}{r}\right) - J\left(\frac{\delta}{r}\right)^3 (3\sin^2\theta - 1)\right\} \quad .. \quad (102)$$

where J is a small quantity involving the oblateness of the Earth,

$\mu = GM$, G being the universal gravitational constant and M being the mass of the Earth,

r = the radial distance from the geocentre,

θ = the geocentric latitude,

δ = the equatorial radius.

In the case where $J = 0$ the basic equations of motion become, in any set of geocentric, rectangular, space-fixed axes,

$$\ddot{x} + \frac{\mu x}{r^3} = 0 \quad .. \quad .. \quad .. \quad (103)$$

with similar equations in y and z. The solution is a conic, with one focus at the centre of the Earth, and six constants of integration are involved. The set of constants employed in the first paper[1] were as follows:

a = the semi major axis of the conic,

e = the eccentricity,

σ = the value of the Mean Anomaly at a particular passage of Apogee.

Ω = the longitude of the ascending node on the equator relative to a space-fixed reference direction in the equator.

ω = the longitude of apogee in the plane of the conic from the ascending node and measured in the direction of motion,

θ_m = the maximum geocentric latitude attained by the particle, *i.e.*, the inclination of the orbit.

In the case of an initially parabolic orbit, $e = 1$ and a is infinite. To obtain a solution for the parabolic case it is therefore necessary to choose constants other than a and e which remain finite for a parabola. In this paper two such constants are used:

$p = a(1 - e)$ the distance, at perigee, from the geocentre,

$q = a(1 - e^2)$ the semi latus rectum of the conic.

These elements remain finite for a parabola, $e = 1$. The equation of the conic then becomes

$$r = \frac{pq}{p + (q-p)\cos f} \quad .. \quad .. \quad (104)$$

where f is the true anomaly, and the complete set of elements is p, q, σ, ω, Ω, θ_m.

III. PERTURBATIONS IN THE GRAVITATIONAL POTENTIAL

Consider now the perturbations, Ψ', in the Gravitational Potential where:

$$\Psi = \frac{\mu}{r} + \Psi' \quad .. \quad .. \quad (42)$$

The form of the particle coordinates can be assumed to be unchanged although the six elements are no longer constant.

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† The de Havilland Aircraft Company, Welkin House, Charterhouse Square, London, E.C.1.; formerly with de Havilland Propellers Ltd.

Denoting the elements p, q, σ by α_i $(i = 1, 2, 3)$ and Ω, ω, θ_m by β_i $(i = 1, 2, 3)$ respectively, it can be shown[3] that the time derivatives of the elements are given by;

$$\sum_{i=1}^{3}[\alpha_j,\alpha_i]\dot{\alpha}_i + \sum_{i=1}^{3}[\alpha_j,\beta_i]\dot{\beta}_i = \frac{\partial \Psi'}{\partial \alpha_j} \quad (j = 1,2,3)$$

$$\sum_{i=1}^{3}[\beta_j,\alpha_i]\dot{\alpha}_i + \sum_{i=1}^{3}[\beta_j,\beta_i]\dot{\beta}_i = \frac{\partial \Psi'}{\partial \beta_i} \quad (j = 1,2,3)$$

The term $[\alpha_j,\alpha_i]$ denotes a Lagrange bracket and has the following properties

$$\left.\begin{array}{l}[\alpha_i,\alpha_j] = -[\alpha_j,\alpha_i] \\ [\alpha_i,\alpha_i] = 0 \\ \dfrac{\partial}{\partial t}[\alpha_i,\alpha_j] = 0\end{array}\right\} (i,j = 1,2,3)$$

Then by the definitions of Lagrange brackets and p, q in terms of a, e:

$$[p,\alpha_i] = [a,\alpha_i]\frac{\partial a}{\partial p} + [e,\alpha_i]\frac{\partial e}{\partial p} \quad (i \neq 1)$$

$$[p,\beta_j] = [a,\beta_j]\frac{\partial a}{\partial p} + [e,\beta_j]\frac{\partial e}{\partial p} \quad (j = 1,2,3)$$

with two similar expressions for

$$[q,\alpha_i]\,(i \neq 2), \quad [q,\beta_j]\,(j = 1,2,3)$$

and using Moulton[4] it follows that:

$$[p,q] = [\omega,\Omega] = [\omega,\theta_m] = [\sigma,\Omega] = [\sigma,\omega] = [\sigma,\theta_m] = [p,\omega] = [p,\Omega] = [p,\theta_m] = [q,\theta_m] = 0 \quad (105)$$

and that:

$$\left.\begin{array}{ll}[q,\omega] = -\tfrac{1}{2}.\dfrac{(\mu q)^{\frac{1}{2}}}{q} & [p,\sigma] = -\dfrac{(\mu)^{\frac{1}{2}}(p-q)}{(2p-q)^{3/2}} \\ [q,\sigma] = -\tfrac{1}{2}\dfrac{(\mu)^{\frac{1}{2}}p}{(2p-q)^{3/2}} & [\Omega,\theta_m] = -(\mu q)^{\frac{1}{2}}\sin\theta_m \\ [q,\Omega] = -\tfrac{1}{2}.\dfrac{(\mu q)^{\frac{1}{2}}}{q}\cos\theta_m & \end{array}\right\} (106)$$

IV. TIME DERIVATIVES OF ELEMENTS

By considering the perturbative accelerations acting on the particle (S,T,W, where S is along the outward radius, T is in the plane of the osculating conic and in the direction of motion, and W is normal to the plane of motion), then on using the equations in the above section:

$$\dot{p} = \frac{pq}{(\mu q)^{\frac{1}{2}}(p-q)}.\left(\frac{p^2}{r} - r\right)T - \frac{p^2}{(\mu q)^{\frac{1}{2}}}.R\sin f \quad (107)$$

$$\dot{q} = \frac{2qr}{(\mu q)^{\frac{1}{2}}}T \quad .. \quad .. \quad .. \quad .. \quad .. \quad (108)$$

$$\dot{\sigma} = -\frac{(2p-q)^{\frac{1}{2}}}{(\mu)^{\frac{1}{2}}(p-q)}\left[\left\{\frac{2(p-q)}{p}r + q\cos f\right\}S - q\left(1+\frac{r}{q}\right)T\sin f\right] \quad (109)$$

$$\dot{\Omega} = \frac{W}{(\mu q)^{\frac{1}{2}}}.r\,\mathrm{cosec}\,\theta_m\sin(\omega+f) \quad .. \quad .. \quad (110)$$

$$\dot{\omega} = -\frac{W}{(\mu q)^{\frac{1}{2}}}r\cot\theta_m\sin(\omega+f) + \frac{q}{(\mu q)^{\frac{1}{2}}(p-q)}\left\{Rp\cos f - p\left(1+\frac{r}{q}\right)T\sin f\right\} \quad .. \quad (111)$$

$$\dot{\theta}_m = \frac{Wr}{(\mu q)^{\frac{1}{2}}}\cos(\omega+f) \quad .. \quad .. \quad .. \quad .. \quad (112)$$

Where $S = 3\mu J\delta^2\dfrac{[3\sin^2\theta_m\sin^2(\omega+f)-1]}{r^4}$

$$T = -3\mu J\delta^2\frac{\sin^2\theta_m\sin 2(\omega+f)}{r^4}$$

$$W = -3\mu J\delta^2\frac{\sin 2\theta_m\sin(\omega+f)}{r^4}$$

Thus the p, q elements give simple expressions for $q\dot{\Omega}$, $\dot{\theta}_m$.

V. EXPRESSIONS FOR ELEMENT PERTURBATIONS

On writing $h = r^2.(df/dt)$, as was done by Ewart[1], and transforming both sides of the six linear differential equations (107)–(112) by $d/dt = (h/r^2).(d/dt)$, the left-hand sides are of the form $d\alpha_i/df$ and $d\beta_i/df$, and on neglecting terms in second and higher orders of J, initial values of h, p, q, ω, θ_m can be used in the right-hand sides, as well as the osculating value of r.

1. Expression for $\Delta\theta_m$

By transforming as above, Equation (112) becomes:

$$\frac{d\theta_m}{df} = -\frac{3}{2}.\frac{J\delta^2}{h_0}\left(\frac{\mu}{q_0}\right)^{\frac{1}{2}}\frac{\sin 2\theta_{m0}\sin 2(\omega_0+f)}{r} \quad (113)$$

On substituting for h_0 and r and integrating:

$$\Delta\theta_m = \frac{3}{4}.\frac{J\delta^2}{q_0^2}\sin 2\theta_{m0}\left[\left(\frac{q_0}{p_0}-1\right)\cos(2\omega_0+f) + \cos 2(\omega_0+f) + \frac{1}{3}\left(\frac{q_0}{p_0}-1\right)\cos(2\omega_0+3f)\right]_{f_0}^{f} \quad .. \quad (114)$$

where $f = f_0$ corresponds to the time origin.

Hence to the first order of J, the orbital inclination of the particle oscillates about the plane of the undisturbed conic.

2. Expression for $\Delta\Omega$

Similarly Equation (110) integrates to give:

$$\Delta\Omega = -\frac{3J\delta^2}{q_0^2}\cos\theta_{m0}\left[f + \left(\frac{q_0}{p}-1\right)\sin f - \tfrac{1}{2}\left(\frac{q_0}{p_0}-1\right)\sin(2\omega_0+f) - \tfrac{1}{2}\sin 2(\omega_0+f) - \tfrac{1}{6}\left(\frac{q_0}{p_0}-1\right)\sin(2\omega_0+3f)\right]_{f_0}^{f} \quad .. \quad .. \quad (115)$$

Thus the line of nodes regresses each orbital period by $(6\pi J\delta^2 \cos\theta_{m0})/q_0^2$ radians.

3. Expression for Δq

On transforming Equation (108), using Equation (113):

$$\frac{dq}{df} = 2q_0 \tan\theta_{m0}\frac{d\theta_m}{df}$$

Hence

$$\Delta q = 2q_0 \tan\theta_{m0}\Delta\theta_m \quad .. \quad .. \quad (116)$$

showing that the semi latus rectum oscillates about its original value.

4. Expression for Δp

Equation (107) can be re-written, using Equation (108) in the form

$$\dot{p} = \frac{p^2}{(\mu q)^{\frac{1}{2}}}\left\{\frac{pq}{p-q}\cdot\frac{T}{r} - S\sin f\right\} - \frac{p}{p-q}\cdot\dot{q}$$

giving after lengthy integration:

$$\begin{aligned}\Delta p = {} & \frac{3J\delta^2 p_0^2}{q_0^3}\times \\ & \times\Bigg[\tfrac{1}{2}(3\sin^2\theta_{m0}-2)\left\{1+\tfrac{1}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\cos f + \\ & + \tfrac{1}{4}(3\sin^2\theta_{m0}-2)\left(\frac{q_0}{p_0}-1\right)\cos 2f + \\ & + \tfrac{1}{24}(3\sin^2\theta_{m0}-2)\left(\frac{q_0}{p_0}-1\right)^2\cos 3f - \\ & - \tfrac{1}{16}\left(\frac{q_0}{p_0}-1\right)^2\sin^2\theta_{m0}\cos(2\omega_0 - f) - \\ & - \tfrac{3}{4}\sin^2\theta_{m0}\left\{1-\tfrac{1}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\cos(2\omega_0+f) - \\ & - \frac{\tfrac{1}{2}\sin^2\theta_{m0}}{\left(\frac{q_0}{p_0}-1\right)}\left\{1+\tfrac{3}{2}\left(\frac{q_0}{p_0}-1\right)^2\right\}\cos(2\omega_0+2f) - \\ & - \tfrac{3}{4}\sin^2\theta_{m0}\left\{1+\tfrac{1}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\cos(2\omega_0+3f) - \\ & - \tfrac{3}{8}\sin^2\theta_{m0}\left(\frac{q_0}{p_0}-1\right)\cos(2\omega_0+4f) - \\ & - \tfrac{1}{16}\left(\frac{q_0}{p_0}-1\right)^2\sin^2\theta_{m0}\cos(2\omega_0+5f)\Bigg]_{f_0}^{f} + \\ & + \tfrac{1}{2}\cdot\frac{\Delta q}{\left(\frac{q_0}{p^0}-1\right)} \quad .. \quad (117)\end{aligned}$$

where $\Delta q = 2q_0 \tan\theta_{m0}\,\Delta\theta_m$.

Thus the Δp expression contains no secular terms.

5. Expression for $\Delta\sigma$

Lengthy integration of Equation (109) leads to

$$\begin{aligned}\Delta\sigma = {} & \frac{3J\delta^2(2p_0-q_0)^{\frac{1}{2}}}{q_0^{3/2}\left(\frac{q_0}{p_0}-1\right)}\Bigg[\tfrac{1}{2}(2-3\sin^2\theta_{m0})\left(\frac{q_0}{p_0}-1\right)f + \\ & + \tfrac{1}{2}(3\sin^2\theta_{m0}-2)\left\{1-\tfrac{5}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\sin f + \\ & + \tfrac{1}{4}(3\sin^2\theta_{m0}-2)\left(\frac{q_0}{p_0}-1\right)\sin 2f + \\ & + \tfrac{1}{24}(3\sin^2\theta_{m0}-2)\left(\frac{q_0}{p_0}-1\right)^2\sin 3f + \\ & + \tfrac{1}{16}\sin^2\theta_{m0}\left(\frac{q_0}{p_0}-1\right)^2\sin(2\omega_0 - f) + \\ & + \tfrac{1}{4}\left\{1+\frac{17}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\sin^2\theta_{m0}\sin(2\omega_0+f) + \\ & + \tfrac{3}{4}\left(\frac{q_0}{p_0}-1\right)\sin^2\theta_{m0}\times\sin(2\omega_0+2f) - \\ & - \tfrac{1}{12}\sin^2\theta_{m0}\left\{7-\frac{13}{4}\left(\frac{q_0}{p_0}-1\right)^2\right\}\sin(2\omega_0+3f) - \\ & - \tfrac{3}{8}\left(\frac{q_0}{p_0}-1\right)\sin^2\theta_{m0}\sin(2\omega_0+4f) - \\ & - \tfrac{1}{16}\left(\frac{q_0}{p_0}-1\right)^2\sin^2\theta_{m0}\sin(2\omega_0+5f)\Bigg]_{f_0}^{f} \quad .. \quad (118)\end{aligned}$$

From Equation (118) it is apparent that the theory breaks down for an initial hyperbolic orbit, when $2p_0 < q_0$, and that $\Delta\sigma$ is identically zero for an initial parabolic orbit.

Secondly the variation in mean anomaly is purely periodic for an initial elliptic orbit where $\sin^2\theta_{m0} = \frac{2}{3}$.

6. Expression for $\Delta\omega$

Finally the time derivative of the perigee longitude can be written as:

$$\dot{\omega} = -\frac{2rR}{(\mu q)^{\frac{1}{2}}} - \frac{Wr}{(\mu q)^{\frac{1}{2}}}\cot\theta_m\sin(\omega+f) + \frac{p\dot{\sigma}}{q^{\frac{1}{2}}(2p-q)^{\frac{1}{2}}}$$

using Equations (109) and (111).

Hence

$$\begin{aligned}\Delta\omega = {} & -\frac{3J\delta^2}{q_0^2}\Bigg[(4\sin^2\theta_{m0}-3)f + \\ & + (4\sin^2\theta_{m0}-3)\left(\frac{q_0}{p_0}-1\right)\sin f + \\ & + \tfrac{1}{2}(1-4\sin^2\theta_{m0})\sin(2\omega_0+2f) + \\ & + \tfrac{1}{6}(1-4\sin^2\theta_{m0})\left(\frac{q_0}{p_0}-1\right)\sin(2\omega_0+3f) + \\ & + \tfrac{1}{2}\left(\frac{q_0}{p_0}-1\right)(2-5\sin^2\theta_{m0})\sin(2\omega_0+f)\Bigg]_{f_0}^{f} + \\ & + \frac{p_0}{q_0^{\frac{1}{2}}(2p_0-q_0)^{\frac{1}{2}}}\Delta\sigma \quad .. \quad .. \quad .. \quad (119)\end{aligned}$$

It will be noted that $\Delta\omega$ is finite for an initial parabolic orbit and also that $\Delta\omega$ is purely periodic when the particle is launched in a plane with inclination given by $\sin^2\theta_{m0} = \frac{4}{5}$.

Further, all coefficients remain finite, in the above equations, for an initial parabolic orbit and the solutions to the six linear equations in the element time derivatives have been obtained.

VI. CONCLUSION

By selection a new pair of elements, $p = a(1 - e)$ and $q = a(1 - e^2)$, in place of the usual conic elements a and e, the solution of the equations of motion of a particle, moving under gravitational forces only, about an oblate spheroid may be developed in a form applicable to motion in initially parabolic or nearly parabolic orbits. It may be noted that to calculate the time of flight the methods of Part II of this paper[2] must again be adopted.

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LEGAL LIABILITY IN SPACE: AN ENGLISH VIEW*

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ABSTRACT

Reviews the position, according to English law, as regards legal liability for damage occurring (*a*) from crashes of space vehicles or satellites on Earth, and (*b*) collisions or mishaps in space, and suggests the standards to be adopted.

I. INTRODUCTION

THE purpose of this article is to consider whether, according to English Law, there should be strict liability in space operations, or whether some other legal standard should be applied.

As yet no damage on Earth has been sustained, so far as can be ascertained, from a space operation such as would give a claim in damages to the injured party. But as activities increase, not to mention international rivalry in racing to be the first to achieve a given goal, the chances of such damage occurring become larger from day to day. This is particularly so when considering the recent successes achieved in re-entry techniques, since a space vehicle or satellite may the better be able to penetrate the Earth's atmosphere when completing its journey or life-span and so crash somewhere upon Earth with possible injury to life and property.

It would seem that this will be the main field of liability in the future and the most productive of legal claims, and so falls to be dealt with first. The other main field to be discussed later will be collisions and other mishaps occurring in space between manned or unmanned vehicles, matters of prime importance to the operators but of less concern to the majority of persons on Earth.

II. DAMAGE ON EARTH FROM SPACE OPERATIONS

1. TWO VIEWS ON LIABILITY

There would seem to be two main viewpoints in relation to legal liability for space operations generally *vis-à-vis* other states and individuals. One is the principle of strict liability for the escape of dangerous or noxious matter as in the leading case of *Rylands* v. *Fletcher*.[1] This involves the principle that a man (or by analogy, a state) who has on his land anything likely to do mischief if it escapes, must keep it in at his peril and is *prima facie* liable for all natural consequences of its escape (or launching, in this context).

Defences have included consent or default of the plaintiff, hardly applicable here; an act of a stranger or third party, possibly applicable; Act of God, hard to establish in an operation so novel and hazardous as spaceflight, since the English legal definition is "an operation of natural forces so unexpected that no human foresight or skill could be reasonably expected to anticipate it."[2] It might be said that since so little is still known about space anything likely to befall a spaceship is within the comprehension of a competent astronautical engineer. A similar defence in tort, inevitable accident, is not available under the rule in *Rylands* v. *Fletcher*.

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2. Nuisance and Negligence

It can be contemplated that many cases of damage may fall not only within the broad law of trespass to land, but also in the realm of *nuisance*. Though broader in scope than the rule in *Rylands* v. *Fletcher*, the law of nuisance is not however so strict in liability, avoiding as it does the acts of an independent contractor, and being confined mainly to injuries affecting the use and enjoyment of land. Even no knowledge of a defect has been held to be a defence.[3]

A line of reasoning more to the point has been adopted in relation to what are described as *dangerous chattels*. Here the action is based primarily in negligence, but in a leading English case Lord MacMillan described the duty as "a degree of diligence so stringent as to amount practically to a guarantee of safety."[4] In this context the duty of care would be almost tantamount to strict liability.

3. Results of the Doctrine

The application of such a doctrine would have to mean that states felt that space operations were so inherently dangerous and uncontrollable that the highest possible duty of care was to be expected. As spaceflight is still in its infancy this idea is understandable, and it is doubtful if complaint would be made if this standard were applied to nuclear energy projects which are infinitely more uncontrollable and potentially destructive. As techniques improve, however, it is foreseeable that states will not impose a standard so strict, for it will mean that the launching state will be liable for all direct and natural consequences of the acts complained of which are not too remote, and this can open up a wide field of liability, including loss of profits by a corporation or even shock in an individual.[5]

This view of strict liability would certainly achieve a measure of quick justice so far as the injured party was concerned, liability being agreed by all parties in such a case, and the matter being taken up by the state whose subject he is, since an individual cannot prosecute a claim himself against a foreign sovereign state in international law.[6]

If the spacefaring nations would agree to the claims being heard in the courts of and according to the *lex situs* of the injury, facility would be achieved, or, if matters had to be adjudicated upon the international level, the machinery of the International Court of Justice at the Hague is established and suitable for such claims, being somewhat preferable to arbitration where a number of similar actions are likely, since *ad hoc* tribunals, like arbitrations, do not have the ability to build up case law precedent in the same way as can a permanent court, a factor which is obviously desirable in a new field of law.

4. Alternative Standard of Liability

There seems to be, however, a second heading in tort under which this liability could be classified. The law of negligence supplies a compendious title for the acts which fall to be considered. Broadly, the application of the law of negligence would tend to impose a lesser, though real, duty of care upon the spacefaring state more in accord with the operation of motor vehicles than aircraft. While under the Air Navigation Act, 1920, no action will lie in respect of trespass to land by private aircraft flying over it, if damage ensues, liability is complete without proof of negligence. Even at common law, the liability would seem to be stricter than in the case of motor vehicles where negligence must always be proved.

5. The Law of Negligence

Negligence is the breach of a duty to take care resulting in damage to the plaintiff. A duty must exist in law, not merely a moral or religious duty, though all persons who are reasonably likely to be affected by the acts or omissions can be considered as legal "neighbours" for this purpose.[7] The doctrine of *res ipsa loquitur*, under which the law assumes that in the ordinary course of events the given accident does not happen is unlikely to find much favour in a field of such uncertain quantities.

It will be seen that if the defendant has behaved reasonably and discharged all the burden of duty which a prudent and skilful person could do, he will not be liable. He may even have a defence of *necessity*, unavoidable action to save himself, his property or to protect others,[8] though in such cases he may have to make restitution for the damage he has done, a practice followed in some American States. Thus the liability in law is far from strict or absolute here, while still requiring considerable obligation on the part of the operator, though there is authority for the proposition that there are really no things dangerous in themselves, but merely some requiring more care than others.[9]

6. Results of the Doctrine

If liability is established, however, the damages will include all direct and natural consequences of the act, unless they are too remote. This applies to both principles outlined above and is ultimately a question of reasonable foreseeability on the facts of the case.[10] It must be emphasized, however, that whichever of these two views is adopted, the defendant will still be able to avail himself of all the defences which English law allows, such as contributory negligence or default on the part of the plaintiff, and also Act of God, necessity or inevitable accident already mentioned.

On balance, while it seems equitable that strict liability should apply in relation to persons and property on Earth, the ordinary law of negligence allied to a high duty of care is more legally realistic for the future, and is to be preferred by this writer.

III. Other Space Vehicles

The second head of liability is that which affects vehicles and operations in space in relation to other vehicles and crews. Here both parties are undertaking

similar risks, and while no one would suggest that consent, the doctrine of *volenti non fit injuria*, would negative liability for damage or injury, clearly the injured party is not in exactly the same position as a person on Earth unconnected with space activities. It would seem that the doctrine of strict liability imposes too harsh a standard here, and thus a duty of reasonable care in the law of negligence is both fairer and more legally justifiable. All persons embarking on spaceflight will be expected to be in possession of the highest qualifications, as incompetent crew would surely be evidence of negligence in a launching state as much as defective equipment. A duty would thus be owed as before to "all persons who are so closely and directly affected by my acts that I ought reasonably to have them in contemplation when I am directing my mind to the acts or omissions which are called in question."[11] Since all activities in space are likely to be carried out by governments rather than by private concerns, claims for injury to individuals would have to be taken up by their parent state. All usual defences will equally be open to a defendant sued for an accident in space, and if he (the captain or operator) has acted with that degree of skill and care expected, he will not be liable, and the loss or damage will have become another hazard of spaceflight.

It has been suggested that the hazards involved should be provided for by an international guaranty fund[12] contributed to by interested states—a solution which would greatly mitigate possible hardships in cases where the defendant state has been not liable in law. Clearly the loss should not fall entirely upon the innocent party, whether he be another spacefarer or a person or persons upon Earth.

IV. Criminal Liability

A survey of liability would not be complete without some consideration of criminal liability, including acts done intentionally. So far as individuals are concerned, acts which occur on space vehicles and which are contrary to the laws of the parent state will be punishable by that state, while acts which are internationally criminal committed against other vehicles or persons and property on Earth will presumably be punishable by the state which apprehends such persons. Since the Nuremberg trials, crimes against peace are now a recognized field of personal liability, and the doctrine of *respondeat superior* should not avail a guilty spacecrew. A master is not liable for the individual acts of his servants not authorized by him and which require an element of *mens rea*, or guilty intent, though where the act is expressly forbidden by statute (or in this case, international law or treaty) he *may* nevertheless be liable if the criminal acts of the servants can be regarded as the acts of the master, even where the act has been against orders or without the master's knowledge.[13] A servant will not be liable if the act alleged was not manifestly unlawful,[14] but he cannot shelter behind his master if the acts complained of were obviously illegal and criminal, but an "Act of State" authorized or subsequently ratified by the government may afford a defence to a crew captain to certain acts done against foreign persons or property.[15]

Responsibility for authorization of crime or indeed any act in space can be assured by insisting that all spacecraft carry the flag of a state to whom resort can be had for redress in cases of injury. Such a state will be accountable according to the machinery of international law today.[16]

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PROPOSALS FOR A PRESSURIZED CREW ENVIRONMENT CONDITIONING SYSTEM FOR A MANNED ORBITAL VEHICLE*

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(Communication from P. Frankenstein & Sons (Manchester) Ltd.)

ABSTRACT

Proposals are advanced for a pressurized suit for use during flights of relatively short duration in the upper atmosphere, and for the associated equipment (gas flow and supply system, helmet, suit suspension system, etc.). Preliminary consideration is given to the design of such equipment for a more advanced missile-type vehicle capable of a limited number of Earth orbital circuits.

I. INTRODUCTION

It has long been recognized that the advancement of science and development of present-day engineering techniques will enable men of Earth to venture into regions on the limits of and beyond the Earth's atmosphere in a comparatively short time.

To enable journeys of this nature to be attempted, the financial backing behind such projects, and the enthusiasm necessary to see successful accomplishment, will have to be considerable. Viewing the situation from a nationalistic point of view, the position of a British venture of such a nature can only be considered in terms of a very slender budget, and consequently a prime target in the design of a British system would be financial economy.

From a technical viewpoint, it was the interest of the authors and their associates in the field of astronautics and their desire to be associated with the practical work involved, that led, through work closely allied with the present paper, to the suggestion that a cheap, light and safe system is potentially available. It was also the hope of the authors that from the effort and concentration put into systems for aircrews to enable acts of war to be carried through, a system might also be evolved to preserve human crews during excursions of a more civilized nature.

The following proposals are based on the known potential requirement for an integrated system of environment conditioning. The equipment would be capable of maintaining a human occupant in an operationally functionable condition during preliminary and subsequently prolonged flights. Flight paths of Earth-orbital and sub-orbital pattern are considered to be the type which would be undertaken in the early stages of accomplishing manned satellite vehicle operation.

The design study takes into consideration the necessity of intensive mass saving. Relative financial economy with consequent early production availability is also envisaged.

System proposals cover the operating of two stages of vehicle development: firstly, the more conventional type of airframe, which could perform flights of relatively short duration in the upper layers of the Earth's atmosphere (Fig. 1), and secondly, the more advanced type of structure capable of performances such as a limited number of Earth orbital circuits (Fig. 2).

The first type of vehicle envisaged is a design system similar to the known North American X-15 project. It is possible that similar projects are in mind elsewhere, and that at some stage a human "payload" would be included. Later stages could be of either the aeroplane-type or missile-type vehicle, but whatever the design of the structure or its foreseen flight path, the proposed crew system, based on the same proposals, could be integrated.

The concept was formulated with no desire to be dogmatic. In many instances the proposed methods or approach to certain problems leave much to be desired. However, the basic proposals can still remain firm together with the presentation of an available skeleton system in the form of practical functioning hardware.

The proposals in the detail designs aim at a unit make-up based upon existing production techniques. The underlying purpose is to provide a system of known general performance characteristics yet capable of direct development into the "spacesuit" type of concept.

The proposals of the general concept suggest a crew atmospheric environment space of minimal size. This is accomplished by dispensing with a pressurized cabin and substituting a pressurizable suit. Suit operating differential pressures in the region of 5–5½ lb./in.² are proposed, and these pressures would produce substantially rigid suit fabric outer structures. Positioning of the occupant within the airframe structure is accomplished by the provision of a harness assembly fitted over the suit envelope and being then directly attached to suspension points distributed around the vehicle cockpit structure.

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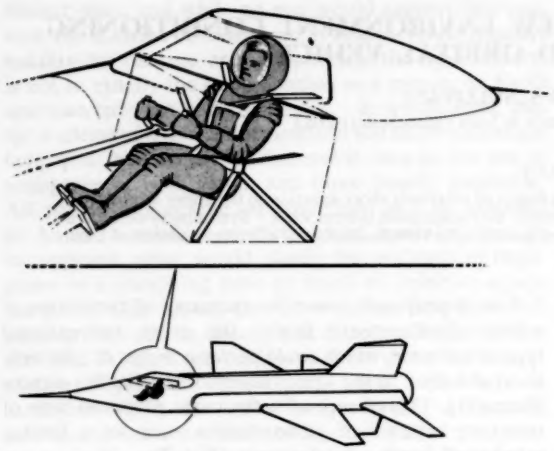

FIG. 1. Envisaged aeroplane-type vehicle (Scheme 1) with pilot position.

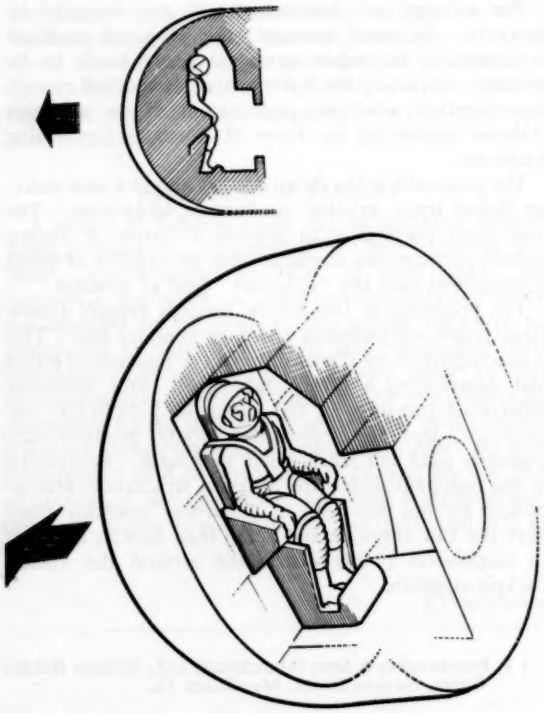

FIG. 2. Envisaged missile-type vehicle (Scheme 2) with pilot position.

Advantages claimed for such an arrangement are the weight economy by dispensing with both the cabin pressurization and sealing equipment, and the conventional metallic seat structure. Greater efficiency in the physiological support of the body of the occupant is suggested by the wider and more numerous distribution of the retainer strap locations. Maintenance and servicing could be considerably eased by the increase of available working space within the cockpit, particularly during the flight stage servicing almost right up to take-off of the vehicle.

Pressurization systems capable of controlling the interior pressure conditions of a suit should be smaller and lighter than the equipment necessary to control a full cabin. The engineering of the mechanisms would not necessarily be simpler, however.

Heating and ventilating would be performed by a closed cycle, relatively small volumetric capacity, system which assumes the suit to be a sealed container. The use of a single gas supply for both breathing and ventilating would also be possible.

Physiological protection against "*g*" loadings in a vertical component could be provided by the combining of a type of anti-*g* suit garment with the lower portion of the pressure suit assembly. Inflation of this suit would also be by oxygen and wastage of gas would be prevented by dumping the exhaust volume into the pressure suit gaseous system.

It is likely that temperature control problems will be considerable. High temperatures are particularly likely to be present, owing to aerodynamic kinetic heating, this being more so in earlier flights than later ones.

The avoidance of pressurizing the cabin would provide a good thermal barrier between the heated vehicle

TABLE I

Scheme 1. (*Aeroplane Type Vehicle*)	
Duration:	Up to 3 hr.
Flight plan:	Air-launched, ascent, descent on skip technique, glide landing.
Position:	Conventional sitting—(limited posterior/anterior accelerations. Vertical accelerations requiring *g* protection).
Equipment:	Full pressure suit with ventilated suit and *g* protection—Flotation gear—(prolonged zero-*g* unlikely).
Required pilot operating capacity:	Conventional flight controls and vision requirement.
Environment:	30/35,000 ft. Suit conditions on demand oxygen at positive mask pressure.
Scheme 2. (*Missile-Type Vehicle*).	
Duration:	Up to 48 hr.
Flight plan:	Ground-launched/orbital/recoverable landing.
Position:	Vertical, (prolonged posterior/anterior, acceleration and deceleration). No vertical *g* protection necessitated.
Equipment:	As Scheme 1. In-flight feeding and urination. *g* massage.
Required pilot operating capacity:	Minimal movement.
Environment:	25,000 ft. Suit conditions on re-breathing. Prolonged zero-*g* likely.

structure and the suit and human occupant. Advantage may also be taken of this insulating characteristic by providing a venturi-induced evacuation of the cabin airspace which would prolong the time of minimum heat transfer from the structure to the suit down to quite low altitudes.

Hazards of having a large 100% oxygen cabin atmosphere would be avoided and the provision of a 100% oxygen breathing mixture would be ensured without the risk of dilution by mask leak.

The two schemes are briefly outlined in Table I.

II. SCHEME 1

1. General Considerations

The Stage 1 proposals consider the possible requirements in an early type of high altitude research vehicle, the airframe design being of more-or-less conventional pattern. Total continuous flight endurances are taken at being not more than three hours. The suit structure is based on a working pressure differential of 4·5–5 lb./in.² with vertical positive "*g*" protection being included. This figure provides suit conditions of 28,000/30,000 ft. relative pressure equivalents at ambient altitudes in excess of 100,000 ft. Helmet proposals take into consideration two schools of thought. One is that the pilot's head and neck should be left unencumbered and free to move within the pressure helmet. The second is that the head and neck are encased in a close fitting pressure headpiece. Either of the two configurations are capable of integration with the general system design.

2. The Pressurization System (*see* Fig. 3)

The pressurization system assumes in effect a gastight sealed suit. The pressurizing gas is oxygen fed from the vehicle liquid oxygen storage system at a gas pressure of approximately 70 lb./in.². The gas is fed into an airframe-mounted regulator which provides a top-up flow into the suit envelope and maintains the suit differential pressure at the required amount.

Over-pressurization, as caused by vigorous suit movement or by exhausting the mask breathing volume into the suit, is restricted by a dump valve governed at some 0·5 lb./in.² above suit selected pressure.

Selection of suit differential working pressure would be manual, selections of full (5 lb./in.²), half pressure (2·5 lb./in.²), or depressurization being available. It might be required to mount the manual selector on the suit, *e.g.*, in the right upper leg position. Alternatively, the pressure differential of the suit could be selected on the airframe-mounted regulator with an emergency depressurizing control or dump valve on the suit. This would avoid additional piping or control runs from the suit to the relatively remote regulator.

Pressurization of the Stage 1 proposals takes into consideration the use of two regulator units.

One is the main suit pressure monitoring unit, and is fed with oxygen at about 70 lb./in.². This regulator produces an output supply, as determined by the

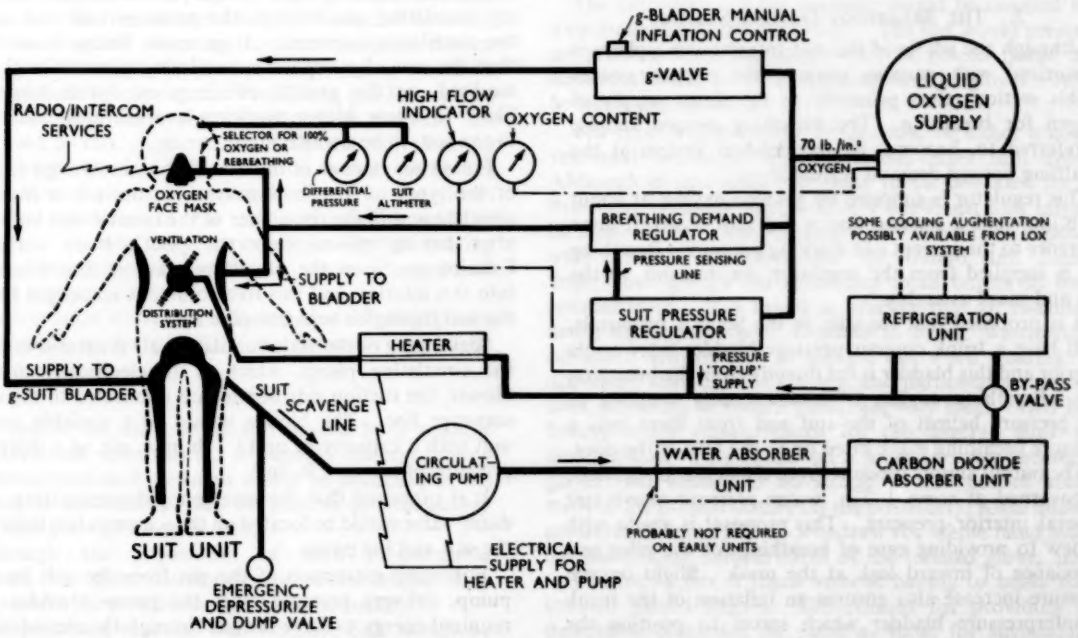

Fig. 3. Schematic view of gas flow/supply system.

selector, feeding into the closed circuit ventilating system by a top-up flow or dump according to the limiting pressure fluctuations of the general system.

The second regulator of the system is really the demand breathing regulator of the oxygen supply, from which the output pressure is governed via a pressure sensing line from the main monitoring regulator.

This then provides an appropriate breathing oxygen supply at some 1–2 in. water pressure in excess of the ventilating system. The prime supply for the breathing regulator is from the main 70 lb./in.^{2} feed system from the liquid oxygen storage unit.

It would be advantageous to provide a "high flow" warning indicator on the feed pipe between the pressure monitoring regulator and the general closed circuit system. Although an existing unit of this type may not be immediately available, an indicator working upon the lines of a differential pressure gauge may be capable of development. The instrument could function by means of a venturi choke being inserted into the main regulator output pipeline. A tapping from the choke area compared across an aneroid with the pressure in the open bore supply pipe could well indicate visually or audibly by the aneroid attached mechanism as high flow rates occurred.

Indications of suit conditions, primarily on visual instruments, would be a necessity and suit interior relative altitude and working differential pressure would require to be indicated. This could be accomplished by existing types of conventional instrumentation but the remote indication of the liquid oxygen storage contents may present a problem.

3. The Breathing Oxygen System

Although the whole of the suit interior atmosphere is pressurized with gaseous oxygen, the oxygen system in this section refers primarily to the direct supply of oxygen for breathing. The breathing oxygen supply, as referred to, becomes an independent system at the breathing oxygen demand regulator.

This regulator is supplied by gaseous oxygen at about 70 lb./in.^{2} and a sensing line is fed into it which gives reference to the current suit working pressure. Breathing gas is supplied from the regulator, on demand, to the suit and mask assembly.

It is proposed that the suit, in the Stage 1 proposals, shall have a trunk counter pressure bladder fitted to its interior and this bladder is fed directly from the breathing line. Breathing oxygen is simultaneously supplied to the pressure helmet of the suit and from there into a pressure breathing mask fitted on to the face of the pilot.

The output supply pressure from the demand regulator is governed at some 1–2 in. water pressure above suit general interior pressure. This proposal is made with a view to providing ease of breathing for the pilot and avoidance of inward leak at the mask. Slight oxygen pressure increase also ensures an inflation of the trunk counterpressure bladder which serves to position the pilot's body, and cushion it, inside the suit envelope.

The demand regulator could have fitted to it a selector which would provide high pressure breathing supplies for use in the case of suit leaks or pressurization control malfunction.

The fit of the oxygen mask on to the pilot's face would have to be good, and an automatic tensioning device, the type of which is already in existence, could be used on to the installation to advantage.

4. The Ventilation and Temperature Control System

Temperature control and ventilation of the suit interior is carried out by a closed cycle gas flow system.

Ventilating gas is supplied to the suit through a single hose of approximately $\frac{7}{8}$in. inside diameter. Distribution of the ventilating gas within the suit could be done by one of two methods: (*a*) by the use of a simple five-point distribution system, or (*b*) through a relatively complex piping system fitted to a separate ventilated suit worn under the pressure suit garment.

Advantages of the five-point distribution system are that it can be fitted to the interior surface of the pressure suit fabric and it does not require a separate garment to be worn. Its distribution is not as efficient as a separate-suit complex pipe system, however.

Use of a separate ventilated suit would provide a ventilation gas distribution over the body near to the physiological ideal but with the necesisty of donning a separate garment prior to fitting the full pressure suit.

Extra bulk and weight is an additional penalty, and there is the necessity for a further pipe connection carrying ventilating gas through the pressure wall and into the ventilating garment. It is most likely, however, that the use of a separate ventilating garment will be required, as the greater efficiency of distribution is likely to allow higher working temperatures, exterior to the suit, to be withstood.

Following passage of the ventilation air through either of the types of distribution systems, the gas is free to circulate within the remainder of the interior suit volume after having passed over the pilot's body surface. Exhaled gas from the breathing mask is also released into this interior, and this free volume is scavenged from the suit through a separate pipe hose.

Scavenging of the stale ventilating air is carried out by the circulating pump, which is an electrically driven blower, the suction side of which is connected to the suit scavenge line. The blower would be a variable speed unit with a capacity of up to 2 lb./min. air, at a delivery pressure of about 1 lb./in.^{2}.

It is proposed that the emergency depressurizing and dump valve would be located on the scavenge line between the suit and the pump.

Following extraction of the gas from the suit by the pump, delivery pressure from the pump provides the required energy to force the gas through the reconditioning circuit.

Four separate units would be situated on the closed circuit line through which the ventilating gas would have to pass prior to re-entry into the suit Of these units, three would be controlled either manually or automatically and could be in operation for intermittent periods, and the fourth unit would be in operation continuously.

The unit in continuous operation would be the carbon dioxide unit and it is proposed that absorption be carried out by filtration through lithium hydroxide.

It is possible that early systems could dispense with the moisture absorber unit owing to the limited time of operation when considered against the moisture output from the body into an initially very dry system.

However, it is proposed that relative humidities in excess of 35% should be avoided. Water removal would be carried out by chemical absorption with lithium chloride or magnesium perchlorate or by freezing-out in the refrigeration unit.

Having removed the waste products from the ventilating gas, the gas would either pass through or by-pass the refrigeration unit or heater as required.

It is more probable that cooling of the gas will be required for a greater part of the flight time than heating, and some augmentation of the refrigeration requirements may be obtained from the liquid oxygen storage system evaporation circuit. General refrigeration could be carried out by a liquid nitrogen refrigeration system or alone by utilizing the low temperatures associated with the liquid oxygen system. Pressurization of the closed cycle system would be carried out by injection of gas in sufficient quantities to provide the required pressure, and it is proposed that this "top-up" supply would flow into the system between the conditioning units and the suit.

In the case of operating conditions which require heating of the pilot, the general body heating would be accomplished by passing the ventilating gas through an electrical heater just prior to entry into the suit. In cases such as this, augmentation of the extremity heating would be required and this could be carried out by the installation of electrical heating element into the gloves and foot portions of the suit. Control would be through a rheostat switch permitting selected heating of the extremities independently from the general heating system.

Experimental evidence indicated that heating power requirements in excess of 30 W. per extremity is unlikely.

5. Suit Design

To enable the crew to be surrounded by a suitable environmental atmosphere, a capsule for containing the atmosphere and crewman would be produced in the form of a gas-holding suit, or what is in present-day terms called a *Full Pressure Suit*.

Although the production of such pressurizable garments has never been extensive, considerable data exist on which a suitable design could be based.

It is proposed that the Full Pressure Suit would be designed as a single one-piece garment, and that it would consist basically of two layers of fabric, one layer (the inner) to be gas-holding and impermeable, and the outer layer the stress-bearing structure. An outer cover may be required; this would only provide a reflective surface in association with temperature control problems.

The inner fabric of the suit would be made from an impermeable proofed fabric, such as Neoprene-coated Nylon or Terylene. Its function would be to contain the gas under pressure and to form the suit's inside skin, to which piping distribution systems, etc., could be attached. The torso section of the garment would consist of a double layer of gas-holding fabric joined in such a way as to enable the intermediate space to be inflated separately, so providing trunk counter-pressure or cushioning of the body within the main pressure walls of the suit.

The outer fabric of the suit would in essential be the stress-bearing medium of the inflated inner material A secondary function would be to provide protection from wear or puncture of the inner layer. By producing the inner fabric larger in size than the outer, all fabric stress of high inflation loads would be passed from the inner to the outer fabrics. In so doing, the cut of the outer garment would be designed so that the necessary movement of limbs could be obtained by displacing the fabric tensions to given locations. Balancing of the stresses would ensure that comparatively little effort would be required on the part of the wearer to move the limb or suit section to the required position. A number of ways already exist in which this can be done, and some designs have quite considerable efficiency.

The extremities of the garment would be covered by a continuation of the suit fabric. The feet would present no problem and they might well be placed inside an additional boot, but it is proposed that the gloves would be made detachable.

The type of glove to be fitted could be either a conventional five-finger type or a one-finger mitt type. Although at first sight preference might be stated for a five-finger type, the provision of a one-finger mitt might be more useful. The one-finger mitt consists of a glove with separate compartments for thumb and index fingers but groups the remaining three fingers in one compartment. It is found in practice that all essential tasks, and most others, required in flying present-day advanced aircraft can be carried out when wearing a one-finger mitt, and the heating, and particularly ventilation, problems in such a glove, are very much eased by having larger compartments for the fingers. Bulk is also reduced, in particular on a more complex pressure glove. An additional advantage in the use of such mitts in the present concept is the possibility, when under pressure, of being able to withdraw the whole hand into the palm/finger compartment of the inflated glove, and to be able to flex it there without having to move the stressed gloved fingers. The reasons for providing a detachable glove are the same as those for suggesting opening or quickly removable helmets as discussed later.

It is suggested that prior to flights such as the ones envisaged with the suit system under discussion, the maximum amount of comfort and freedom should be afforded the crewman. Although it may be some considerable time prior to take-off that he is dressed in the outer pressure suit, there would be further operations necessary at preflight check, etc. In this case, the extra comfort of not being completely detached from the ground conditions immediately on donning the suit, and that at least hands and head would be left free until the very last minute, would prove advantageous. Attachment of the gloves would be by a self-sealing inner layer and a positive zipper attachment on the outer fabric.

Entry into the suit could be by a number of ways, such as through the back with the suit opening in the form of an opening vertically or horizontally. Sealing of the entry would be by a gas-tight zipper overlaid by a stress-holding zipper. Layouts in this form have already proved practical and efficient.

The necessity of providing the correct clearances and movement axes would require that the suit be individually made to the body size of the selected crewman.

Associated with the main suit structure design is the possibility of providing an overall cover which would fit over the stress-containing Terylene fabric.

This coverall would be capable of containing a thermal insulating layer of material such as foamed polyurethane, which has extremely good performance as a heat barrier. The outer skin of the coverall could be made of a highly reflective material such as Melinex film coated with aluminium. However, the coverall assembly would be separate from the principal pressure suit garment, and is only suggested as one approach to solving some of the temperature control problems.

6. Helmet Design

General

The design proposal for the helmet configuration has two alternatives. Both have advantages and disadvantages, but the prime difference is that one helmet is an open-fitting type which gives considerable freedom of head movement within the pressure wall and with the main structure being mounted on the shoulders. The other design is a close-fitting unit and is mounted in such a way that the helmet weight is borne on the head and neck. This is caused by the necessity of making the helmet movable at the neck.

In either case of helmet design, it is proposed that the helmet itself shall be so designed that the maximum amount of freedom of movement and access to the face will be retained after donning, but before closing and pressurizing. This point was briefly discussed previously in relation to the provision of detachable gloves, and it is suggested that this provision will be an advantage to the wearer. Primarily the aim is to alleviate the total separation of the crewman from the ground environment after dressing, which would allow a considerable number of tasks to be performed before sealing the suit off. Not least of the advantages is the psychological one of permitting free contact with ground crews and proceedings right up to the time of take-off. Although it is agreed that the psychological outlook of the chosen crew will have been approved, the considered advantages of having opening helmets in relation to their construction difficulties will be worthwhile.

The two schools of thought for open-fitting and close-fitting types of helmets differ primarily on the buffet problem.

Although in the open-fitting type helmet the wearer is capable of moving round inside a shell as it were, and the effect of being in a sealed confined helmet is not nearly so great, the close-fitting helmet gives ample protection and support in the case of encountering severe buffet conditions or high-altitude vibrations during flight.

It would be possible to locate the close-fitting helmet with additional positioning media such as is suggested for the suit suspension system, and this would also support the head directly. Alternatively, the open-fit type of helmet gives greater ventilation on the head. The ease of manufacture is much relieved, as no moving pressurized portions of the helmet are required, and the helmet weight is taken on the shoulders, which relieves fatigue of the head and neck. However, should severe vibration conditions be encountered when wearing an open-type helmet, the head could be shaken inside the pressurized container without any restraint being able to be applied to it. In the severest of cases, unconsciousness or death could be so precipitated in the use of an open-fitting helmet, which would, on the other hand, have been retarded by a close-fitting restrained helmet. Due to the division of opinion, the proposal that both types of helmet be produced is made.

The Open-Fitting Helmet

This type of helmet is the easier of the two to manufacture, and would consist of a fairly substantial structure located as a halter over the shoulders. A base ring would be fitted on the top of this structure and to this would be fitted a quick-attached/released helmet bowl. All the main services required for breathing supply, demisting and communications would be supplied through the halter structure. The wearer's head would be well clear of the base ring.

Fitting and testing of the inner helmet and mask assembly are possible without any restriction, and the final fitting of the pressure helmet bowl is very quick and easy. Similarly, removal of the helmet by the wearer is equally as easy and quick. The materials of construction of the helmet can vary, but in the easiest and cheapest form the rigid halter section would be made from resin fibreglass laminate and a plastic (possibly methyl methacrylate) bowl. The joint ring would be light alloy.

The helmet bowl would not be clear all round, but provided with a reflective outer surface and possibly matt black interior surface. Vision range could be provided

as required and glare protection could be afforded by a separate movable glare outer shield. Helmets of this design have already been experimentally operated in the region of 5 lb./in.^{2} differential pressures.

The Close-Fitting Helmet

Of the two types of helmet, the close-fitting helmet will be more costly to produce and be heavier. It is unlikely that the same amount of freedom of movement would be afforded when pressurized, or when open and unpressurized. It is proposed that the helmet shell would also house the breathing mask, microphone and earphone, and all the head cushioning, and that the head would be fitted into a helmet separately, and then the helmet attached to the suit. Considerable problems exist in the production of a pressurizable close-fitting pressure helmet which would give great comfort for long endurance and provide ample movement of the head. The vision visor of the helmet would be movable, as would the glare blinds, in similar manner to the open-fitting helmet. The requirements of producing head movement would also require that the helmet be capable of movement on the shoulders, and this would necessitate suspending the weight of the helmet on the head and neck. Owing to this, fatigue and discomfort of the neck muscles is likely to be considerable.

Demisting problems of the visors of both types of helmet would be similar and could be carried out either by embedded heating elements or by a dry gas flow over the visor. Both types of demisting have proved effective in an ambient temperature of $-40°$ C. and the demisting problem becomes progressively less with rise in ambient temperature conditions.

With reference to the open-fitting type of helmet, the inner helmet would only provide a locating unit for the earphones and attachment for the breathing mask assembly and tensioning device. The breathing mask would be a pressure demand type with integrated microphone, and the microphone switch would be mounted externally on the suit.

7. The Suit Suspension System

One advantage of proposing a pressure garment is the fact that in so doing, the shape that the garment takes provides a ready made form-fitting seat. From this it is but a short step to suggest dispensing with a metallic seat unit with consequent structure weight saving. Further to this is the advantage of providing a cockpit area without the interference of a seat structure and ease of flight preparation, particularly on small vehicles, can be much eased. The suit system must, however, be positioned and fixed in the cockpit somehow, and it is proposed to do this by means of a harness system (Fig. 4).

When pressure suit garments are worn pressurized, the suit fabrics provide a substantially rigid condition, and it is a relatively easy operation to fit a harness of a design similar to a parachute suspension system over the whole suit. This harness could then be located by

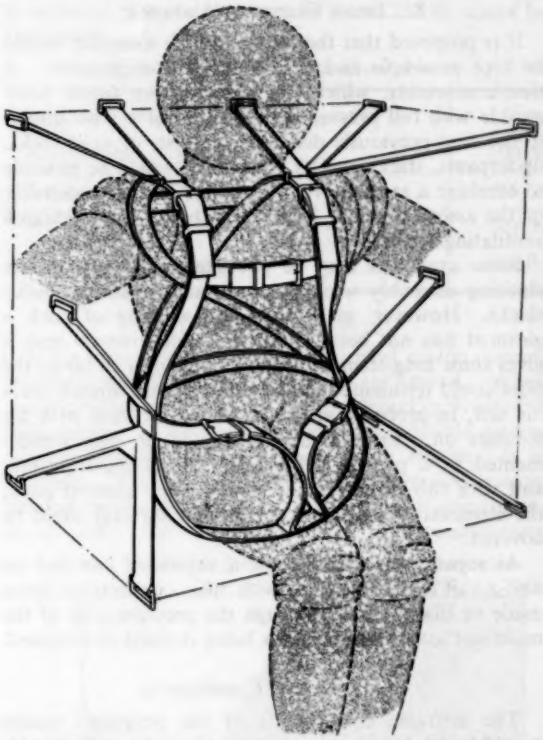

Fig. 4. Schematic view of suspension system. (In practice, quick release units would be used instead of the buckles shown).

harness webs to attachment points on the airframe and by closing the correct location of these points, movement of the suit relative to the cockpit interior structure could be prevented.

Quick-attachment units would be provided at each attachment point probably incorporating a tightening device such as a quick-action turnbuckle for final adjustment. Release of the pilot from the cockpit would be by a single quick-release unit on the suit which would release all the harness on the suit, enabling escape on completion of the flight. It is proposed that a ten-point suspension system would be sufficient (or possibly twelve-point if a close-fitting helmet was being provided). Feet and hands are assumed to be placed and located on conventional flight controls, *i.e.*, rudder bars and control column, and with this attachment it is suggested that the crewman would be substantially located in the cockpit.

Harness materials would be of a type similar to parachute harness webbing, and be capable of withstanding the desired or estimated loads. Resonance of the harness mounted suit unit due to appropriate vibration frequencies of the airframe structure would be possible, but the effect of this could be more easily studied in practice in a rig test rather than being postulated. It is possible that certain vibration effects on the pilot would be even damped by the use of such a system.

8. Inner Clothing Assembly

It is proposed that the inner clothing assembly would be kept as simple and as conventional as possible. A simple assembly, which has already been found compatible with full pressure suit systems of a type similar to the ones previously discussed, consists of wool socks, underpants, slacks and "T" shirt. It would be possible to combine a separate ventilated suit with this assembly or the assembly could be worn with the suit-integrated ventilating system.

Some argument may be put against using an inner clothing assembly which incorporates relatively bulky slacks. However, in practice the wearing of such a garment has not been found to be detrimental and it gives some long-term survival protection in so far as the pilot could terminate the flight and release himself from his suit, in probable alien conditions, at least with his trousers on. External apparel would be only supplemented by a pair of flying boots of standard pattern, and with this assembly and a minimum survival pack, the elements of long-term, post-flight survival could be covered.

As separate undergarments, a ventilated suit and an anti-*g* suit could be worn, with their connections being made or disconnected through the pressure wall of the main suit as the garment was being donned or removed.

9. Airframe Components

The airframe components of the proposed system would be broken down into separate units each capable of being installed independently, as convenient to the airframe construction, and also of being changed or replaced easily, *i.e.*, a true "packaged" system.

The airframe components required would be principally the liquid oxygen system, which would possibly incorporate the ventilating air refrigeration system and/or a separate refrigeration unit, the regulators for breathing supply and suit pressurization control, the circulating pump, the water absorber unit and ventilating gas heater.

Radio would also be incorporated in the airframe but this is regarded as separate from the general system, except for the required lead installation to the pilot's head-set assembly.

In addition, an electrical supply would be required for the pump and heaters at the appropriate rating.

Piping systems connecting the separate components should not be a particular problem, and insulating and lagging of pipes of this type is already well known.

The suit-positioning harness attachment points would also be airframe components but would consist of simple "strong point" structures which would be of minimum bulk and weight.

The Liquid Oxygen System

Concentrating first on Scheme 1, with flight durations of 3 hr. the liquid oxygen system proposed would be a 12-l. system with a component weight of 40 lb.

This unit would provide in the region of 7800 l. gaseous oxygen, and, should the pilot be breathing 100% oxygen independently from the suit interior over a 3-hr. period, his consumption should average about 2100 l. of gaseous oxygen. This should provide a sufficient margin of gas to provide pressure "top-up" or minor leak volume replacement. Should the pilot select rebreathing and not use the direct independent feed, the 12-l. LOX system should be capable of use for up to 36 hr. on a sealed suit system. These approximations are based on a metabolic rate of some 800 ml./min. and it is possible the consumption calculations could be sufficiently well revised to provide a liquid oxygen installation with a more conservative weight estimate.

Ventilating Gas Refrigeration

Refrigeration of the ventilating gas could be provided by utilizing the low temperatures existing in the region of the liquid oxygen system evaporation circuit. It may be found necessary, however, to install a separate unit which could be a liquid nitrogen circuit system with a possible installation weight in the order of 20 lb. Some augmentation to the cooling capacity of this system could also be obtained from the LOX system.

Carbon Dioxide and Water Removal

Removal of carbon dioxide from the ventilating gas could most easily be accomplished by absorption in anhydrous lithium hydroxide. About 1·35 lb. of this compound is needed to absorb 1 lb. carbon dioxide. In assuming a carbon dioxide output of 370 ml./min. per man, the calculated output should be approximately 0·3 lb. carbon dioxide in 3 hr. This would be capable of being absorbed and would ensure a carbon dioxide build-up in the system to be kept well below 1% total volume; a 3% carbon dioxide build-up would be regarded as an upper limit for prolonged exposure.

Combined with carbon dioxide removal could be moisture removal, again by absorption, in lithium chloride or manganese perchlorate, the latter requiring about 2 lb. compound per 1 lb. water absorbed. A combination of the two absorbers may be an advantage. An attempt would be made to keep the relative humidity between 30 and 50% or possibly less, if severe heating problems were likely to be encountered. The installation mass of the combined carbon dioxide/water absorber unit should be in the region of 15 lb.

Heaters and Regulators

Regulator units on the system would be quite small and light and should present no installation problem.

The mass of the oxygen regulators with performance functions comparable to those of the proposed units is 5 lb. and together with instrumentation arrays should not total more than 15 lb. installation mass.

The ventilating gas heater unit would be a small unit compared with the LOX or absorber units, and would be electrically powered and with a possible installation mass of 10 lb.

The circulating pump, also electrically powered, would be capable of circulating up to 2 lb./min. gas flow (possibly up to 40 ft.³/min.) with a pressure rise across the pump of up to 1 lb./in.² It could well be a centrifugal, variable speed, pump, and need not weigh more than 10 lb.

10. Mass Analysis

An analysis of the build-up of the installation mass of the proposed system is based on existing equipment capable of performing comparable tasks. Claims for a mass-saving advantage in the overall proposal are, therefore, based for the most part on known equipment with a generous margin allowed where approximations have had to be made.

Working on this basis, a mass analysis for Scheme 1 proposals can be made with a degree of accuracy at least within the correct order of magnitude (see Table II).

The assumption is that a male crew member will be the operating pilot, with a mass of about 150 lb., and the final all-up mass includes this figure.

The advantages claimed with this proposed system include the dispensing with seat structures, cabin sealing and pressurization, etc., and no estimate of the mass of these structures or components has been made in the final payload/structure mass advantage estimate.

Table II.—*System Mass Breakdown*

Pilot	150 lb.	(actual)
Full pressure suit	15 lb.	(actual)
One 12-l. LOX converter	40 lb.	(actual)
Gas regeneration system:		
CO_2/HO_2 Absorber unit	15 lb.	(estimated)
Heater	10 lb.	(estimated)
Refrigerator unit	20 lb.	(estimated)
Regulators (two)	10 lb.	(actual)
Circulating pump	10 lb.	(estimated)
Suspension harness	6 lb.	(actual)
Instrumentation, piping and insulation	20 lb.	(estimated)
Internal equipment and clothing	8 lb.	(actual)
Total Unit Installation Weight	304 lb.	(including *pilot*)

The tabulation details of the mass breakdown are annotated as to whether the mass is estimated or actual. In the case of the masses quoted as being "actual" equipment, units have been weighed which are parts of the actual first prototype functioning unit which was assembled to prove various statements in the proposals.

In the case of "estimated" quoted masses, masses of units with comparable function have been checked and estimates deduced.

It is possible that during later design stages, unit masses could be still further reduced when integrating them in a final system.

A brief comparison can be made with a similar research vehicle programme equipment assembly (*i.e.*, the North American X–15) and a mass comparison shows an advantage of 176 lb., which includes the pilots in both cases. If an estimated mass for the X–15 seat structure is included, a mass advantage of some 200 lb. could be compared.

Table III.—*X-15 Mass Breakdown*

Air conditioning and pressurizing system	260 lb.
Pilot	220 lb.
Seat structure (estimate)	30 lb.
Total crew/environment system	510 lb.
Present proposal (estimate)	304 lb.
Mass saving advantage	206 lb.

11. Escape/Survival

It has been stated previously that pilot escape systems envisaged with the general proposals are limited. When comparing existing "in-flight" escape data (Fig. 5) with the flight pattern of even early test vehicles, possible success of conventional ejection or bale-out is remote. It is assumed, therefore, that the pilot's airframe escape method would be, at minimum, by capsule separation. In all cases, the complete environment conditioning

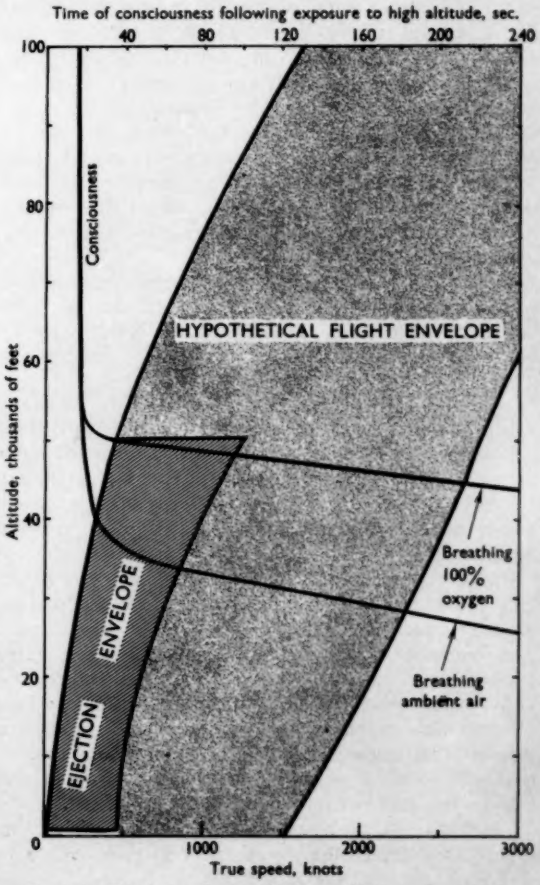

Fig. 5. Ejection seat capabilities *vs.* flight performance.

assembly could be included in the separated capsule which would thus maintain, under most circumstances, normal pilot physiological function.

Assuming, therefore, that the pilot is faced with escape or post-flight abandonment of the vehicle, means should be provided for his long-term survival under alien circumstances.

Quick-release methods have already been discussed in connection with the suspension harness assembly, and this type of release would free the pilot from the capsule/cockpit structures. Helmet release and removal has also been discussed, and this allows freedom and access to Earth surface conditions.

Intact, the pressure suit garment provides the function of an exposure/immersion suit. It is possible that personal flotation could be attached to the suit outer and this assembly has been included in the suit mass estimate.

There is, therefore, means for flotation and survival in the case of a descent into the sea, and with the addition of an elementary dinghy/survival pack, including homing radio beacon, the basis for prolonged sea survival could be provided. Initial tests of full pressure suit flotation and water survival have been carried out, and no critical survival hazard has so far been indicated.

In the case of land survival, particularly in the tropics, means for removing the pressure suit outer garment must be provided. Should it be impractical to permit unaided removal of the suit by the pilot *via* the normal donning entry, means to cut off the suit quite easily must be provided. The inner clothing assembly, including the boots and the emergency pack, would then provide means for prolonged post-flight protection and survival until recovery.

Individual flotation units capable of being incorporated with a suit garment already exist, and the particular design would depend primarily on the selected helmet unit of the suit.

III. Review of Scheme 2

It is accepted that a considerable amount of experience and knowledge would be gained from Scheme 1 units before a Scheme 2 assembly could be operational. However, a number of additions and developments would also be necessary for which proposals can be put forward at this stage.

Primarily the differences are occasioned by the increase in working pressure differential of the suit, and the need for replenishment and refreshment of the crewman.

An increase in pressure differential of the suit working pressure over a much more prolonged operational period would necessitate a more substantial suit. Ease of extremity mobility would be decreased and it is possible that extensible arms with two gloves would be required, so as to permit the use and performance of various functions or operations more easily.

"In-flight" feeding, or to be more accurate "in-flight" fluid replenishment of the crewman, would be provided by a tube attachment to the mask permitting suction of fluid from a store bottle inside, or on the side of, the helmet. In turn, this storage flask could be replenished by capsules capable of being screwed or plugged into a filler point, and then, by squeezing or applying gas pressure to the bottle, provide a flow *via* a self-sealing non-return coupling, thus refilling the storage flask from which the fluid has been drunk.

Urination would be a further problem, and it is proposed that a separate flexible container, filled with absorbent material, could be incorporated in the lower section of the suit. This urination device would, of necessity, have to be put on prior to donning the suit and would be of appropriate capacity. Devices of this nature already exist and could be incorporated into the suit assembly.

The necessity of preventing free urine from being deposited into the suit interior atmosphere is primarily to avoid a rise in humidity with increased danger of visor misting. Secondly, prevention of a decrease in the thermal insulation value of the suit is ensured and additional discomfort is avoided. Methods of faeces absorption have not been propoesd in view of the estimated endurance times of operation.

An additional potential problem which might arise on prolonged voyages whilst wearing one suit would be the beard growth beneath the oro-nasal mask. In extreme cases it is possible that the sealing of the mask to the face may be impaired and that discomfort and irritation might be intense.

A principal mechanical alteration in the Scheme 2 system would be in the oxygen supply. Present-day conventional liquid oxygen systems would malfunction under the envisaged prolonged endurances at zero-*g* and the development of LOX systems capable of functioning at zero-*g* would be necessary.

IV. DISCUSSION

General design and in some cases detail design of the basic proposal has now been set out. It remains to be seen whether there is, in fact, a practical use for such an assembly, and if so, what advantages it presents. In principle, the assembly can be demonstrated with existing equipment, and a number of questions which might have been raised during the theoretical proposal presentation would probably be answerable on the initial test rig.

To further pursue the practical development of the assembly, only a comparatively small financial outlay is required. To progress the unit assembly to the stage of full-scale exhaustive tests on the integrated self-supporting system, additional assistance on the mechanical design side would have to be enlisted. This refers, in particular, to regulator design and supply.

Production time for the general system need not be prolonged, and it could, if necessary, be brought to a prototype flight stage in weeks rather than months or years.

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A RADIOISOTOPE PROPULSION SYSTEM*

By W. SHORT,† Ph.D. and C. SABIN,‡ M.S.

(Communication from Convair (San Diego) Division of General Dynamics Corporation)

ABSTRACT

A propulsion device is studied theoretically which utilizes directly the alpha particles emitted by a radioactive isotope as the exhaust gas. The random direction of emission and the absorption of the decay particles in the solid is considered in the analysis. It is found that 11·8 mg. of ^{232}Th deposited per square centimetre of alpha-absorbing surface will produce 0·0039 dyne of thrust. The application of this system to space propulsion, to attitude control and to extending satellite life times is discussed.

I. INTRODUCTION

THERE are four general categories of propulsion systems for space vehicles which have been used or discussed in the literature.[1] They are (*a*) those that derive energy from chemical combustion in solid and liquid propellent rockets, (*b*) those that use nuclear energy as their source of power, (*c*) those that use electrical or electromagnetic means for acceleration of mass, and finally (*d*) those that use electromagnetic radiation to obtain a propulsive force. The device discussed here employs a nuclear source of energy; however, the efficiency is not limited by energy-transfer processes as in systems which employ separate fuel and working fluid. In the present system the accelerated mass is the high-energy decay products ejected directly from the radioactive fuel. The magnitude and duration of the thrust achieved are computed by simple momentum considerations. Several applications of this device to space propulsion problems are presented.

II. ANALYSIS

A simple propulsion system producing a small thrust can be designed using energetic alpha particles resulting from the decay of radioactive isotopes as the exhaust gas. Just as in the case of conventional jet devices, the thrust of such a system is equal to the rate of change of momentum of the exhaust particles. The force per particle is

$$-F = \frac{dp}{dt} = \frac{d(mv)}{dt} \quad .. \quad .. \quad (1)$$

The kinetic energy rather than the momentum or velocity of particles resulting from natural radioactive decay is given in the literature.[2] The momentum per particle, p, can be determined from the mass and energy of the particle by using the definition of kinetic energy. The relativistic equations for momentum and kinetic energy (E) are

$$p = \frac{m_0 v}{\sqrt{1-\beta^2}} \quad .. \quad .. \quad .. \quad (2)$$

$$E = m_0 c^2 \left(\frac{1}{\sqrt{1-\beta^2}} - 1 \right) \quad .. \quad (3)$$

where $\beta = v/c$ and m_0 is the rest mass, c being the velocity of light. Eliminating velocity from the momentum equation by first solving Equation (3) for velocity gives

$$p = \frac{E}{c}\left(1 + \frac{2m_0c^2}{E}\right)^{1/2} \quad .. \quad (4)$$

The value of $2m_0c^2/E$ for a typical alpha particle energy of 8 MeV. is $7{\cdot}8 \times 10^2$. The constant "one" in Equation (4) is small compared to $7{\cdot}8 \times 10^2$; therefore, the following nonrelativistic equations for momentum and velocity based upon the rest mass can be used with $< 1\%$ error:

$$p = \sqrt{2m_0E} \quad .. \quad .. \quad (5)$$

$$v = \sqrt{\frac{2E}{m_0}} \quad .. \quad .. \quad .. \quad (6)$$

Products of radioactive decay other than alpha particles were considered, such as electrons, positrons, and protons. The masses of these particles including the apparent increase in mass due to relativity were small compared to alpha particles; consequently, the momentum and thrust produced were also smaller than that obtained by alpha decay.

The natural decay of a radioactive nuclide is a random process in both time and direction. The rate at which particles are emitted is random; however, the average decay rate is expressed by

$$\frac{dn}{dt} = \lambda n \quad .. \quad .. \quad .. \quad (7)$$

where the decay constant, λ, has been measured experimentally for most isotopes. When considering a large number of particles the emitted particle velocities can be considered uniformly distributed in all directions. The term *emitted particle* will be used to refer to the alpha particle immediately after being ejected from the nucleus. The velocity of the emitted particle will usually differ from the velocity of the particles escaping from the system due to collisions in the solid.

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To produce a net thrust the velocities, and consequently the momentum, of the particles leaving the device cannot be uniformly distributed in all directions. Alpha particles are absorbed by a few thousandths of an inch of solid material. A thin shield of absorbing material is, therefore, placed on one side of a film of radioactive material to absorb all alpha particles emitted in that direction. A thrust directed toward the absorber is produced by the particles which are free to leave the device in the opposite direction. A schematic drawing of the device is shown in Fig. 1. As indicated in the

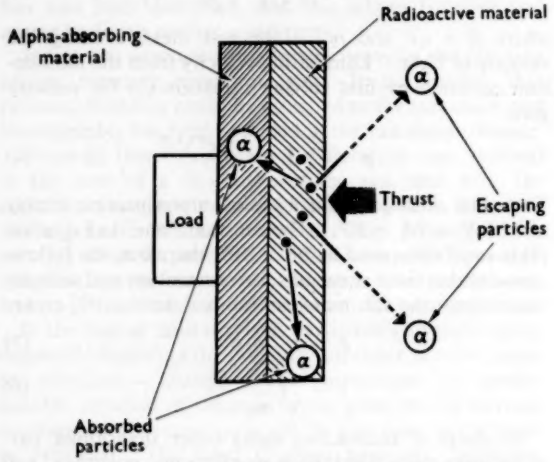

FIG. 1. Schematic of alpha particle propulsion device.

sketch, the alpha particles which are emitted in all directions have three possible dispositions:

(*a*) they can be absorbed in the shield, or

(*b*) they can be absorbed in the radioactive material itself, or

(*c*) the particles may escape from the device and produce thrust.

The energy of absorbed particles is transformed to thermal energy. The velocity distribution of the emerging particles is symmetrical about the x axis. The coordinate system used in the following analysis is shown in Fig. 2. The x component of the particle velocities is the only contribution to the net force on the device. The total force in the positive x direction is

$$-\underline{F}_x = \sum_0^n \frac{dm_0 v_x}{dt} \quad .. \quad .. \quad (8)$$

The net force in the minus x direction will be termed the total thrust, $\underline{T}$. The thrust is related to the x component of momentum of the particles leaving the surface and the rate at which they leave, as shown by Equation (9), which was obtained from Equation (8).

$$\underline{T} = mv_x \frac{dn}{dt} \quad .. \quad .. \quad .. \quad (9)$$

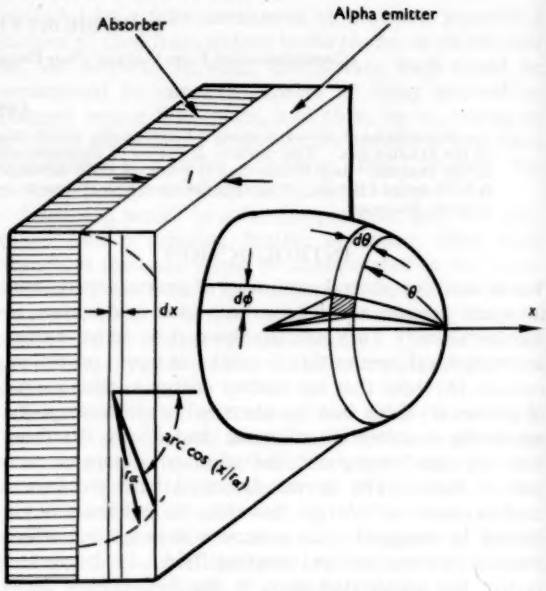

FIG. 2. Coordinate system used to calculate thrust.

Since the alpha particle energies are strongly attenuated by the radioactive substance itself (self-absorption), the number of particles reaching the surface $x = 0$ and their momentum must be determined from an integral over momentum space. This integral is

$$T = \int_0^{x=l} \int_0^{\theta=\frac{\pi}{2}} \int_0^{\phi=2\pi} mv_x \sin\theta \, d\phi \, d\theta \, SAdx \quad (10)$$

The assumptions to be used in this analysis are:

(*i*) All alpha particles emitted in the $-x$ direction are absorbed.

(*ii*) No particles are lost from the edges of the device.

(*iii*) The masses, m, of the particles are uniform.

(*iv*) The area, A, is constant.

(*v*) The particle emission rate, S, is constant.

(*vi*) The absorber and radioactive material are pure substances.

The constant S has the units of particles per second per steradian per cubic centimetre and can be evaluated from the following equation:

$$S = \frac{\lambda \rho N}{4\pi M} \quad .. \quad .. \quad .. \quad (11)$$

Combining the above assumptions with Equation (10) gives

$$T = mS \int_0^{x=l} \int_0^{\theta=\frac{\pi}{2}} \int_0^{\phi=2\pi} v_x \sin\theta d\phi d\theta dx \quad .. \quad (12)$$

The loss of energy by an alpha particle passing through a material is usually presented in the form of a range-energy diagram. The range-energy diagram for alpha particles in a material whose atomic number is 90 is given in Fig. 3. Thorium has an atomic number of 90

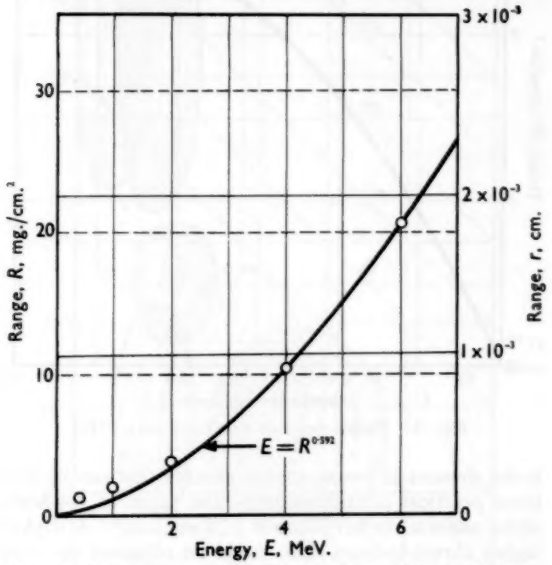

FIG. 3. Range-energy diagram for alpha particles in thorium.

and will be used in the examples cited later. These data were derived from the information given by Friedlander and Kennedy.[2] Range-energy data for alpha-particles can be fitted by an equation of the following form

$$E = R^a = b(\rho r)^a \quad .. \quad .. \quad (13)$$

where $a = 0{\cdot}592$. The range, r, is the distance a particle of initial energy, E, will have travelled when it obtains thermal equilibrium with its surroundings. An alpha particle will travel very nearly in a straight line throughout its range. It is apparent from this that all radioactive surfaces having thicknesses greater than r would have the same thrust.

Combining Equations (6) and (13) gives

$$v = \sqrt{\frac{2b(\rho r)^a}{m_0}} \quad .. \quad .. \quad (14)$$

The velocity of a particle leaving the surface can be determined from its residual range. The residual range is $(r_\alpha - x/\cos\theta)$ where r_α is the range of the emitted particle, x and θ are the location and direction in which the particle was emitted. The velocity of a particle leaving the surface is

$$v_{(x=0)} = \sqrt{2b\rho^a\left(r_\alpha - \frac{x}{\cos\theta}\right)^a \Big/ m_0} \quad (15)$$

his equation can be simplified to give

$$v_{(x=0)} = v_\alpha\left(1 - \frac{x}{r_\alpha \cos\theta}\right)^{a/2} \quad .. \quad (16)$$

The x component of the velocity of a particle leaving the surface is

$$v_x = v_\alpha \cos\theta\left(1 - \frac{x}{r_\alpha \cos\theta}\right)^{a/2} \quad .. \quad (17)$$

If this value of v_x is to be substituted into Equation (12) the limits of integration over θ must be changed. At values of θ greater than arc cos (x/r_α), Equation (17) will predict negative values of v_x, which is not the physical situation. Therefore, the new limits of integration will be $0 > \theta >$ arc cos (x/r_α). Also, the values of x will be made to assume a dimensionless form, X, by dividing by r_α. The thrust can now be computed from the following equation:

$$T = mv_\alpha r_\alpha S\int_0^{X=L} dX \int_0^{\theta=\arccos X}\left(1 - \frac{X}{\cos\theta}\right)^{a/2} \cos\theta \sin\theta\, d\theta \int_0^{\phi=2\pi} d\phi \quad (18)$$

The dimensionless average value of v_x for all particles emitted at X is given by

$$\frac{\bar{v}_x}{v_\alpha} = \frac{1}{4\pi}\int_0^{\theta=\arccos X}\left(1 - \frac{X}{\cos\theta}\right)^{a/2} \cos\theta \sin\theta\, d\theta \int_0^{\phi=2\pi} d\phi \quad .. \quad (19)$$

Integrating ϕ gives

$$\frac{\bar{v}_x}{v} = \tfrac{1}{2}\int_0^{\theta=\arccos X}\left(1 - \frac{X}{\cos\theta}\right)^{a/2} \cos\theta \sin\theta\, d\theta \quad .. \quad .. \quad (20)$$

Equation (20) was integrated using the binomial expansion for $(1 - X/\cos\theta)^{a/2}$. The results for thorium, *i.e.*, with $a = 0{\cdot}592$, are

$$\frac{\bar{v}_x}{v_\alpha} = \tfrac{1}{4}(1 - X) \quad .. \quad .. \quad (21)$$

This equation shows that one-quarter of the momentum of particles originating at the surface, $x = 0$, produces thrust, and at $X = 1$, $(x = r_\alpha)$ none of the particles produce thrust.

If mv_α is considered a measure of the maximum thrust theoretically available from the "fuel," and also if $m\bar{v}_x$ is the actual thrust produced, then $\bar{v}_x/v_\alpha$ is the efficiency of the device. It is evident from Equation (21) that the efficiency per unit mass of fuel decreases linearly from a value of 0·25 at the surface to a value of 0 at the maximum range from the surface, $X = 1$.

Substituting the expression for $\bar{v}_x/v_\alpha$ into Equation (18) yields

$$T = mv_\alpha r_\alpha S\pi \int_0^{X=L} (1 - X)dX \quad .. \quad .. \quad (22)$$

The thrust per unit area for a substance of molecular weight 90 is obtained by integrating Equation (22)

$$T = mv_\alpha r_\alpha S\pi L(1 - \tfrac{1}{2}L) \quad .. \quad (23)$$

Thorium-228 was selected for fuel in the proposed propulsion device. This substance was chosen because it will produce the maximum thrust over a reasonable period of time. This is a result of the following facts:

(1) Thorium-228 emits alpha particles. The advantage of using alpha particles as the propellent was discussed earlier.

(2) For each thorium atom that decays, five alpha particles are emitted in rapid succession. Thorium-228 is also called RdTh, which is an intermediate in the ^{232}Th decay chain. Thorium-232 has a 10^{10} half-life and is unsuitable for the present system. Thorium-228 decays by alpha and beta emission through the isotopes ^{224}Ra, ^{220}Rn, ^{216}Po, ^{212}Pb, ^{212}Bi, and ^{212}Po to stable ^{208}Pb. The longest half-life of these intermediate elements is 3·64 days.

(3) The half-life of ^{228}Th is 1·90 years which would be adequate for most space missions.

The alpha particles emitted by the ^{228}Th chain have the following properties:

Average energy, MeV.	6·4
Velocity, cm./sec.	$1{\cdot}76 \times 10^9$
Specific impulse, sec.	$1{\cdot}79 \times 10^6$
Range R_α, mg./cm.^{2}	23
Range r_α, cm.	$2{\cdot}12 \times 10^{-3}$

The above value of specific impulse is based upon the mass and velocity of the exhaust gases as the specific impulse is defined for chemical fuels. A more realistic figure which is based upon the mass of the fuel ^{228}Th is $1{\cdot}57 \times 10^5$ sec. The thrust of pure ^{228}Th can be computed from Equation (23). When the properties of the fuel are substituted in this equation, the following expression is obtained:

$$T = 1{\cdot}02 \times 10^{-2}L(1 - \tfrac{1}{2}L)\text{dyne/cm.}^2 \quad .. \quad (24)$$

The maximum thrust is obtained when $L = 1$. This thrust is 0·0051 dyne/cm.^{2} Equation (24) is plotted in Fig. 4.

To determine the maximum acceleration obtainable with this device, the ratio of thrust to mass must be maximized. The mass of the device is the sum of the masses of the ^{228}Th and the absorber. In this analysis, the mass of the absorber is that required to stop all alpha particles emitted in the $-x$ direction. The thrust-to-mass ratio using beryllium as an absorber is

$$\frac{T}{W_{\text{fuel}} + W_{\text{absorber}}} = \frac{1{\cdot}02 \times 10^{-2}L(1 - \tfrac{1}{2}L)}{0{\cdot}23L + 0{\cdot}00628} \quad (25)$$

Elements of low atomic number provide the greatest absorption of alpha particles on a mass basis. Beryllium

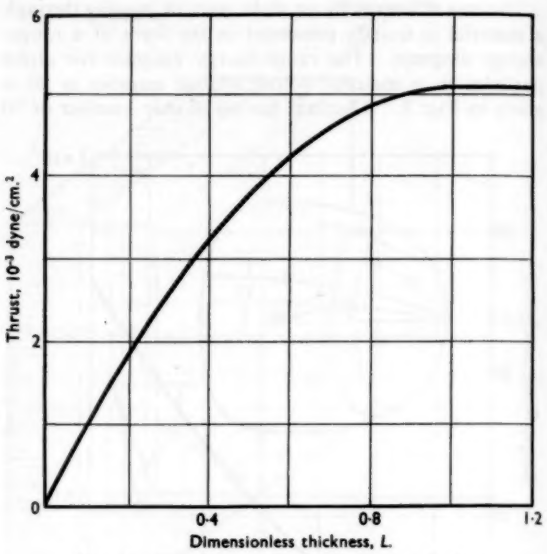

FIG. 4. Thrust per unit area from pure ^{228}Th.

is the element of lowest atomic number that can be used from practical considerations. The range of 6·4-MeV. alpha particles in beryllium is 6·28 mg./cm.^{2} A slightly higher thrust-to-mass ratio could be obtained by using a thinner absorber which would permit a few alpha particles to leave the device in the forward direction ($-x$ direction) at very low energies. The analysis of this case would require solving an equation similar to Equation (10) applied to a two-phase system. Since the mass of the absorber is small compared to the mass of ^{228}Th this complicated analysis was eliminated by assumption (*i*).

The optimum mass of ^{228}Th is found by differentiating Equation (25) and equating to zero. The optimum conditions are

$$\begin{aligned} L &= 0{\cdot}515, \\ T &= 0{\cdot}0039 \text{ dyne/cm.}^2 \\ W_{\text{fuel}} &= 11{\cdot}8 \text{ mg./cm.}^2, \text{ and} \\ T/W_{\text{total}} &= 0{\cdot}216 \text{ cm./sec.}^2 \end{aligned}$$

The maximum acceleration, T/W_{total}, of this device is $2{\cdot}2 \times 10^{-4}\,g$. The acceleration and specific impulse of this device are compared to other propulsion systems in Fig. 5 (see Sutton[1]).

III. APPLICATIONS

The inherent simplicity and reliability of this propulsion mechanism will make it suitable for a wide variety of vehicles requiring a low-thrust propulsion device. It appears to be particularly applicable where it is not desirable to furnish power supplies and controls for the power plant itself, or where low vibrations or a high resistance to damage by meteors are advantageous.

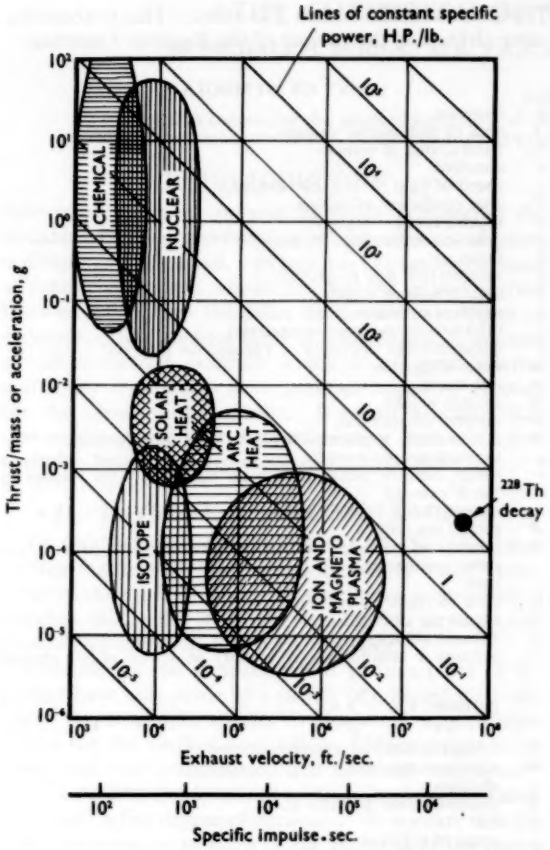

FIG. 5. Regions of accelerations and specific powers for various rockets.

The suitability of this mechanism for several applications is demonstrated in the following examples.

1. Low-Thrust Spacecraft

Most of the usual conclusions about high-specific-impulse, low-thrust spacecraft apply to vehicles using this thrust mechanism. It is apparent that variation in vehicle mass with time is negligible and the exponential decay in thrust is not as great a handicap as it might seem to be. Consider for example, a vehicle with a constant acceleration, f, of $10^{-4}\,g$, accelerating to escape velocity with the thrust directed tangent to the flight path. This problem has been solved.[3] The result for the time to reach parabolic velocity V_i is

$$t_E = f^{-1}(V_i - 8{\cdot}81 \times 10^3 f^{1/4}) \qquad .. \qquad (26)$$

If V_i is assumed to be 25×10^3 ft./sec., then

$$t_E = 6{\cdot}8 \times 10^6 \text{ sec.} = 79 \text{ days.}$$

During this time a ^{228}Th source would have decayed to 92% of its original value.

The optimum weight of ^{228}Th thrust-surface produces approximately 39 dynes/m.^{2} and without payload has an acceleration of $2{\cdot}2 \times 10^{-4}\,g$. Assuming that $10^{-4}\,g$ is an acceptable acceleration for a vehicle and that the payload required is 50 lb., then the total vehicle mass is 91·6 lb., and the thrust required is $91{\cdot}6 \times 10^{-4}$ lb. The area of the alpha-emitting surface is 104·5 m.^{2}. This is a disk 11·5 m. in diameter, or about 37·5 ft. One possible configuration for a vehicle with this performance is shown in Fig. 6. Steering could be accomplished by

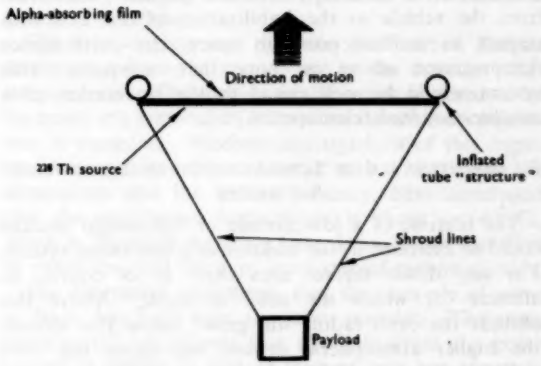

FIG. 6.

shortening or lengthening shroud lines to shift the thrust vector out of line with the centre of mass. Even a large surface of this type is relatively immune to meteor damage, since punctures have only a local effect.

2. Attitude Control for Heavy Astronautical Vehicles

This thrust mechanism appears to be ideal for attitude control of heavy satellites and has a number of obvious advantages over small, intermittent rockets. By the use of gimballed or shuttered surfaces complete attitude control could be accomplished with only a few square feet of the α emitter. Consider for example the rotation of a 10,000-lb. vehicle about a transverse axis as shown in Fig. 7. It is desired to start this vehicle from rest, rotate it through 180°, and bring it to rest again. If I is the moment of inertia of the vehicle and F the force produced at each emitting surface, the angle of rotation

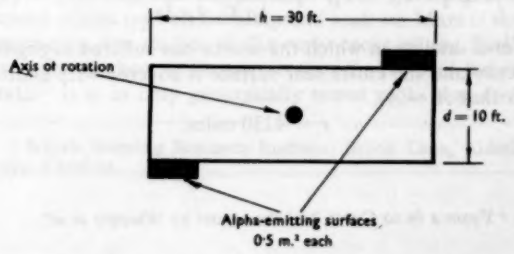

FIG. 7.

is given by

$$\gamma = \tfrac{1}{2}\frac{Fh}{2I}t^2 \qquad .. \quad .. \quad (27)$$

Assuming that the mass is evenly distributed and that the time to accelerate equals time to decelerate;

$$t_{\text{total}} = 2\frac{2\gamma I}{mg} = 1{\cdot}37 \times 10^4 \text{ sec.} = 3{\cdot}8 \text{ hr.}$$

Therefore, 3·8 hr. are required to invert the vehicle. Since most of the attitude control required for a vehicle consists in the removal of residual angular momentum from the vehicle or the stabilization of one axis with respect to a fixed point in space, the performance demonstrated above is more than adequate. This system would be well suited to the orientation of a satellite-mounted telescope.

3. Maintenance of Low-Altitude or Lightweight Satellites

The lifetime of a low-altitude or lightweight satellite could be extended by the addition of a low-thrust system. For any thrust surface area there is, of course, an altitude for which the orbit is stable. Above this altitude the orbit radius will grow; below this altitude the higher atmospheric density will cause the orbit radius to shrink. With the proper choice of initial altitude and thrust the orbit of the satellite could be made to grow slowly with time and then begin to shrink as the strength of the source decreases. The altitude at which the changeover occurs will be that at which the drag force and the thrust are equal

$$T_i A e^{-\lambda t} = \tfrac{1}{2}C_D A_f \rho V^2 \qquad .. \quad (28)$$

This stable altitude will thus be that altitude at which

$$\rho = \frac{T_i A e^{-\lambda t}}{\tfrac{1}{2}C_D A_f V^2} \qquad .. \quad .. \quad (29)$$

Since in a circular orbit the velocity is given by

$$V^2 = \frac{r_0^2 g_0}{r} \qquad .. \quad .. \quad .. \quad (30)$$

and the density of the upper atmosphere is approximately*

$$\rho = \rho_z \exp[-\kappa(r - r_z)^{0{\cdot}4}] \qquad .. \quad (31)$$

an equation may be written for the stable altitude

$$\frac{1}{r}\exp\,[-\kappa(r - r_z)^{0{\cdot}4}] = \frac{T_i A}{\tfrac{1}{2}\rho_z C_D A_f r_0^2 g_0}\, e^{-\lambda t} \qquad .. \quad (32)$$

For a satellite in which the source has suffered negligible decay and the entire rear surface is covered with emitter so that $A = A_f$,

$$r = 4230 \text{ miles.}$$

The stable altitude is then 230 miles. This is about the same altitude as the perigee of the *Explorer I* satellite.

LIST OF SYMBOLS

- a constant.
- A area of radioactive surface.
- A_f frontal area of vehicle.
- b constant.
- c speed of light $= 3 \times 10^{10}$ cm./sec.
- C_D drag coefficient of vehicle.
- E kinetic energy $= \tfrac{1}{2}mv^2$.
- f vehicle acceleration, ft./sec.^{2}
- F force.
- g acceleration due to gravity.
- h axial length of vehicle.
- I moment of inertia.
- l thickness of radioactive material.
- L dimensionless thickness of radioactive material $= l/r_\alpha$.
- m mass of particle.
- m_0 rest mass.
- M atomic weight.
- n number of particles.
- N Avogadro's number $= 6{\cdot}023 \times 10^{23}$ atom/g.-mole.
- p momentum per particle.
- r range, cm., or radius of vehicle trajectory with respect to Earth's centre.
- r_z datum radius $= 4063$ miles.
- R range, mg./cm^2.
- S number of particles emitted per steradian, per second, per cubic centimetre.
- t time.
- t_E time to reach escape velocity.
- T thrust per unit area.
- v exhaust velocity.
- $\bar{v}_x$ average x component of velocity at $x = 0$ of all emitted particles, $\bar{v}_x = \frac{1}{n}\sum_0^n v_x$
- V orbital velocity of a satellite, ft./sec.
- W mass per unit area.
- x distance.
- X dimensionless distance x/r_α.
- β v/c.
- γ angle of rotation.
- θ angle, see Fig. 2.
- κ constant in atmospheric density relation, 1·315 (miles)$^{0{\cdot}4}$.
- λ decay constant, emitted particles per second per total number of atoms.
- ϕ angle, see Fig. 2.
- ρ density.
- ρ_z density at datum radius, $3{\cdot}27 \times 10^{-8}$ lb./ft.^{3}

Subscripts

- — total.
- 0 condition at Earth's surface.
- i initial conditions.
- x x component at $x = 0$.
- α emitted alpha particle.

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* From a fit to Curve 2 of the report by Whipple *et al*[4].

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THE CHEMISTRY OF MARS. II. THE SURFACE*

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ABSTRACT

Information concerning the physical conditions at the Martian surface are reviewed and the various hypotheses as to its chemical nature are discussed. None is considered to give a satisfactory explanation of all the available information.

I. INTRODUCTION

THE information at present available concerning the possible chemical constitution of the surface of Mars is disappointingly small. In fact it is impossible to make a definite statement about any of the many surface features, other than the polar caps, with any degree of certainty. Nevertheless there is a considerable amount of circumstantial evidence which is of great interest and many hypotheses have been advanced to account for the observational results. It seems probable that new evidence will soon be forthcoming that will allow a choice between the present conflicting views.

II. PHYSICAL SURFACE CONDITIONS

To establish the chemical nature of the prominent surface features of Mars without a fairly exact knowledge of the prevailing physical conditions is obviously a hopeless task. But it is only recently that any accurate physical information has been available.

Estimates of the temperature of various parts of the surface have been made by a variety of different methods. Early thermocouple studies[1,2] gave only approximate values for the surface as a whole. More recent work using water-cell transmission and infra-red techniques[3,4] has established the localized surface temperature variations with a fair degree of accuracy. It appears that the night temperature of most of the planet is in the region of − 70° C. The sunlit side, however, is much warmer. On the equator at noon the temperature of the light surface areas reached about 20° C., while temperatures of 33° C. have been recorded for the dark areas.[4]

These upper temperature extremes are similar to many terrestrial values, though the night temperatures are probably lower than anything experienced on Earth. The overall temperature range of 90–100° C. must cause a considerable daily expansion and contraction of the upper solid surface. It is therefore not surprising that polarization studies[5] have shown the surface to be very dusty, probably similar to the Moon's surface. Other studies[6] have confirmed the presence of fine grains.

The modern physical picture of the Martian surface is of crumbled rocks exposed to wide temperature variations. These variations must be also the cause of strong winds, and dust clouds and storms cannot be rare.

III. THE POLAR CAPS

That the white caps of Mars melt and re-form seasonally has been known for a long time and from the start there have been only two rival hypotheses of the chemical nature of the caps.

The first suggests that they are solid carbon dioxide precipitated from the atmosphere by the extreme polar temperatures. The melting of the cap is assumed to be the result of a seasonal increase when the critical temperature is exceeded. Modern investigations of this hypothesis[7] considering the amount of carbon dioxide in the atmosphere and the known pressure, have concluded that the equilibrium temperature is about − 144° C., which is far below anything recorded. It can therefore be concluded that the caps are not solid carbon dioxide.

The alternative hypothesis, that the caps are of snow or frost, has long been the more popular. The same investigator[7] has shown that the light from the polar regions is similar to that of frost at very low temperatures (*i.e.*, about − 70° C.) and it seems likely that the caps are regions of thinly deposited ice crystals. That they are not deep snow drifts similar to the polar regions of the Earth is established by the speed with which they melt and disappear.

It is probable that the poles are the only parts of the Martian surface to ever have surface water or ice.

IV. THE LIGHT SURFACE MARKINGS

The reddish markings of Mars have often been called deserts and this is probably close to the truth. That they are dusty and covered with fine grains was discussed above, but the chemical composition of these grains remains an unsolved problem.

Polarization studies[8] of the deserts led to the conclusion that they have properties similar to terrestrial limonite, $Fe_2O_3 \cdot nH_2O$. This is a fairly common mineral on Earth which is variously named "bog-iron ore," "brown hematite," etc. It is a hydrated ferric oxide and is yellow-brown in colour.

The possibility of the presence of an oxidized compound of this type on a widespread scale on Mars is very interesting, for studies of the early rocks of the Earth[9] have shown them to contain iron in the reduced ferrous state. It is in only geologically recent rocks that ferric

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iron can be found. These observations of terrestrial minerals are used as evidence that the primitive atmosphere must have been reducing and only fairly recently changed to its present oxidizing form. On Mars, however, which has no free oxygen in its atmosphere, the presence of limonite would indicate that at some time in the past an oxidising atmosphere was present.

Suggestions to account for the formation of widespread oxidized materials have been made[10] by making the assumption that Mars once possessed atmospheric oxygen. By the usual action of solar ultra-violet radiation it can be shown that the ozone layer would form much nearer to the surface than the Earth's. Interaction between the layer and the surface might then have occurred readily.

However, it cannot be definitely concluded that the deserts are of limonite. Other studies of the areas[11] have shown that the spectral distribution of their reflected light closely resembles that of felsitic rhyolite: a mineral mixture of quartz, SiO_2, and alkaline silicates, *e.g.*, $KAlSi_3O_8$. Such minerals provide no insight into the nature of the atmosphere.

Supporting evidence for the silicate hypothesis of the deserts comes from studies[12] of quite different phenomena. Radiations from the deserts do not conform to the Stefan-Bolzmann Law, but the deviations can be partially explained if the areas are selectively reflecting silicates.

V. THE DARK SURFACE MARKINGS

The dark areas of the Martian surface were for a long time assumed to be large bodies of open water similar to the seas of the Earth. But although this is now known not to be true the areas are still known as maria. One of their most interesting properties is a change of colour from green and blue to brown and red and the reverse. Antoniadi[13] has catalogued the various reported colour changes.

Largely from considerations of these changes, early observers[14,15] of Mars suggested that the maria might be areas of vegetation. At first it was assumed that the colour changes were exactly seasonal, but as many people have now pointed out[16] this is not quite true and the colours are never exactly in accord with the prevailing Martian season as would be expected from the vegetation hypothesis.

Another objection to the vegetation explanation is the lack of oxygen in the atmosphere from plant photosynthesis. This can be explained in a number of ways.

One suggestion[17] is that under the rigorous conditions of Mars, plants retain their oxygen within their tissues for use in respiration during non-photosynthetic periods. Such an explanation, however, is not very satisfactory, for most terrestrial plants produce by photosynthesis far more oxygen than they require. A more likely suggestion to account for the oxygen deficiency is that of Kuiper[18] who proposes that the areas may be lichens; simple plants with only a very low oxygen output. Such oxygen as is evolved into the atmosphere will be lost from the planet or removed by chemical combination with reduced surface materials at a rate faster than it is being produced.

But the presence of vegetation implies the presence of chlorophyll, or some similar porphyrin photocatalyst. Chlorophyll has characteristic reflection properties, reflecting red and green wavelengths well, but other colours badly. Studies of the reflection properties of the maria[19] have established that they reflect only blue and violet wavelengths.

Considering the differences between Mars and Earth, it is not surprising that possible vegetation may have developed different properties in response to the climate and physical conditions. Tikhov[20] has shown that terrestrial blue vegetation from high altitudes will make more efficient use of sunlight in conditions of extreme cold than will green. This is a possible explanation of the colour of the Martian areas. The same worker[21] has apparently demonstrated that vegetation from high altitudes reflects red and infra-red radiations to a much lower extent than the normal.

It would therefore seem to be possible to explain the observed properties of the Martian maria in terms of the properties of terrestrial vegetation from conditions not dissimilar to those prevailing on Mars. Yet direct evidence of the existence of vegetation, or of life of any kind has not yet been produced.

Interesting observations supporting the vegetation hypothesis have been published recently.[22] Comparisons of infra-red spectra of the maria have revealed the presence of absorptions due to carbon-hydrogen bonds, which are the fundamental units of organic molecules. It is possible, however, that the molecules so detected are simple compounds of abiogenic origin.

Other objections to the presence of vegetation on Mars have been raised. The previously discussed temperature differences between the maria and the desert has been used as an argument for the inorganic nature of both.[23] The observed difference of about 13–15° C. is approximately the value that can be predicted from Boltzmann's law assuming the albedos of the regions to be 0·30 for the deserts and 0·15 for the maria. It is argued that if the maria are regions of vegetation they must absorb radiation differently from inorganic materials and the temperature difference should not be predictable by a simple physical law.

In the absence of any adequate information about the radiation and reflection from various types of vegetation and bearing in mind the probable selective radiation[12] of even the deserts, it is questionable whether this objection can at present be serious.

Other explanations of the colours of the maria in terms of simple inorganic phenomena have been proposed. The secondary spectra of refracting telescopes is one suggestion[24] but is obviously invalid for the published results[16] indicate that the observed colours are the reverse of those predicted.

A strong alternative hypothesis of the maria is the volcanic ash suggestion of McLaughlin[25,26] The proposed scheme explains the observed surface features in completely non-biological terms. By maintaining as a basic supposition the existence at various points of the Martian surface of active volcanoes, McLaughlin suggests that streams of dust and ash will be thrown into the atmosphere. This material will be carried by winds for long distances and then eventually deposited on the surface in patterns that will depend upon the surface wind distributions.

The colour changes can be explained in terms of different moisture contents of the ash at different seasons. Many minerals found in terrestrial volcanic ash occur in a variety of colours depending upon the degree of hydration.

Two objections can be made to this volcanic hypothesis. The first is Öpik's observation[27] that the surface features of Mars possess powerful regenerative abilities that are not associated with inorganic materials.

The second objection, due to Kuiper,[28] maintains that the water supply on Mars is insufficient to produce the observed extremes of colour change. Answering this objection McLaughlin[26] has suggested that the changes may be due to water released by vegetation. In this case plant life is a primary, but unseen and indirect, cause of the observed properties of the maria.

VI. THE CANALS

The remaining surface features of Mars are the much-debated canals. In his reviews of many years of observation, Lowell[29,30] has claimed to see the canals as planet-wide surface constructions of intelligent beings. But in view of the difficulties of observing the canals and establishing them to have the form described by Lowell, it is very uncertain whether they are to be regarded as optical illusions or actual surface markings. Webb[31] has recently attempted to revive Lowell's hypothesis and claims to have demonstrated that the maps of the canals follow a pattern that is quite dissimilar to any known interconnecting system found in nature, such as ground cracks or insect webs.

At the present time the canals remain an open question and their chemical significance, if they exist, is quite unknown.

VII. CONCLUSIONS

The chemical nature of the surface of Mars remains a mystery. The various hypotheses that have been discussed above each explain a number of selected features, but none can be said to be a satisfactory explanation of all the available information.

With the recently developed techniques and interest in extraterrestrial chemistry, it seems likely that the present intensified studies will produce valuable results.

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NEWS AND ANNOUNCEMENTS

B.I.S. NEWS

Lecture Programme

The Lecture Programme for the first half of the 1961 session was distributed with the October issue of the *Journal;* it discloses that as far as meetings are concerned this is probably the most intense period of activity so far in the Society's history. It will include no less than four symposia (one international and one industrial) with the period of six months, and more detailed announcements concerning some of these appear below.

We hope that as many members as possible will take advantage of this programme. We would remind all members that they are entitled to bring guests and colleagues to meetings, and these will be particularly welcome at the industrial symposium.

Lecture notices announcing these meetings have been prepared and it is hoped that they will be displayed as widely as possible. Those who can display posters or assist in other ways (*e.g.*, by including notes in staff or college magazines) are earnestly invited to do so. The posters and additional copies of the lecture programme are obtainable on request from the Secretary, who will be pleased to provide any further help or information.

Course on Rocket Motor Technology

The Society is organizing a one-day course on Rocket Motor Technology, intended primarily for teachers, it being felt that the average teacher's lack of knowledge concerning rocket propulsion is one of the reasons why this subject does not receive the attention it deserves in present curricula (thus it usually receives little, if any, mention in courses on heat engines, which are often mainly concerned with the obsolescent steam engine).

The course will be given at the College of Advanced Technology, Birmingham, on Thursday, **30 March, 1961**. Details of the programme will be made available later; it is hoped that it will be possible to include demonstrations of propellents and rocket engines. Further particulars are obtainable from the Secretary.

Symposium on Liquid Hydrogen

A one-day symposium on the Use of Liquid Hydrogen as a Rocket Propellent is being organized by the Society and will be held in London on Friday, **28 April, 1961**.

The provisional programme includes the following papers:

Liquid Hydrogen Production on an Industrial Scale, by T. J. Webster and K. C. Smith (British Oxygen Research and Development Ltd.).

Liquid Hydrogen as a working Fluid in Advanced Propulsion Systems by P. A. E. Stewart (Hawker Siddeley Aviation Ltd.).

Use of Hydrogen in Nuclear Rockets, by Dr. O. H. Wyatt (Hawker Siddeley Nuclear Power Co. Ltd.).

Use of Liquid Hydrogen in Large Rocket Engines, by A. V. Cleaver (Rolls-Royce, Ltd.).

Design of Rocket Engines Burning Liquid Hydrogen as a Fuel, by D. Hurden (The de Havilland Aircraft Co. Ltd.).

Further particulars of the venue and programme will be made available later; members and others who are considering attending the symposium are invited to notify the Secretary of their intention.

Symposium on the Communications Satellite

A one-day symposium on the use of the Communications Satellite is being organized by the British Interplanetary Society and will be held in the Council Room of the Federation of British Industries, 21, Tothill Street, London, S.W.1, on Friday, **12 May, 1961**, from 10 a.m. to 5 p.m.

About six papers will be presented for discussion, including the following:

Communications Satellite Systems Suitable for Commonwealth Telecommunications, by Dr. W. F. Hilton (Hawker Siddeley Aviation Ltd.)

An attempt to assess the importance of language differences, peak traffic periods, distribution of population and of telephones on the design of satellites and their orbits. An active repeater in an elliptical orbit with apogee at 63° N. is compared with other communications systems.

The Synchronous Communication Satellite, by R. P. Haviland (General Electric Company, U.S.A.).

Examines the characteristics of the synchronous satellite when used in communications service and concludes that the special problem of time delay will cause lower altitudes to be preferred for real-time telephone communication services, but that the synchronous height is useable for all other services, and is strongly indicated for broadcast services. Considers the requirements for this service, and discusses short-term and long-term solutions to design problems.

Low-Cost Communications Satellites, by A. M. Peduzzi (Aerospace Consultant).

The high cost of satellite systems proposed at present results from the high cost of the booster itself, in development, production and launching, owing to its large size. In order to decrease the costs of satellite payloads it is necessary to miniaturize the payload and hence the booster. The proposed use of Scout to orbit a 150-lb. communications satellite with voice and TV capability illustrates how this miniaturization can take place today. The purpose of the proposed paper is to show how further miniaturization and refined rocketry can decrease the system's size and cost to about one-quarter of that of the Scout system, resulting now in an economic and rational concept of British space operations inside a limited budget.

The Engineering and Economics of Satellite Communication Systems, by G. K. C. Pardoe (The de Havilland Aircraft Co. Ltd.).

The basic rocket engineering problems of establishing a system of Earth satellites to provide world-wide communication facilities are discussed. The potential utilization of such a system is explored, an analysis of the economics of the project being included. Ground-support equipment associated with the launching arrangements are examined, together with the satellites and their associated ground facilities.

Active Communications Satellites, by Dr. G. E. Mueller (Associate Director, Space Technology Laboratories). (Summary not yet available).

The Courier Satellite, by G. F. Senn and P. W. Siglin (U.S. Army Signals Research and Development Laboratory). (Summary not yet available).

Details of further papers to be presented will be made available later. Members and others who are likely to attend the symposium should inform the Secretary (British Interplanetary Society, 12, Bessborough Gardens, London, S.W.1) as soon as possible.

Second Space Medicine Symposium

Plans are now being drawn up for holding the Second Space Medicine Symposium and members or others who think they may wish to present papers are requested to inform the Secretary. The symposium will not be held before Autumn 1961.

Proposed Australian Branches

Some time ago a circular was sent to approximately ninety B.I.S. members in Australia asking whether they would be interested in the formation of an Australian Branch and whether they would be prepared to help with its administration and/or contribute towards its finances.

A favourable response to the circular having been received, Council has decided to proceed a stage further. Several of the replies received from Australia made the point that most of the membership is in three groups in three different states, and therefore it is proposed that three branches should be established—in Adelaide, Melbourne and Sydney.

A second circular has now been sent to Australian members suggesting that preliminary meetings should be convened in these areas, to elect branch committees and officials and to discuss future policy. In accordance with our usual practice, if the members do decide to set up branches, these will operate initially as informal groups until they can be regarded as established on a permanent basis.

Members in other parts of the Commonwealth are reminded that the Council is always willing to receive proposals for the establishment of new branches. Two of the conditions which must obviously be satisfied before such proposals can be approved are that (*a*) there should already exist within a convenient distance of the proposed centre sufficient members (20–30) to form the nucleus of a branch, and (*b*) there should be sufficient and suitable members available to undertake the administration of the proposed branch. The Council and Officers of the Society will give guidance and assistance to the branches, but their continued existence and further development must always be mainly dependent upon the enthusiasm and efficiency displayed at branch level.

Derby Informal Group

The address of Mr. D. A. Wigley, Secretary of the Derby Informal Group, is now Four Winds, Burley Drive, Quarndon, Derby.

Compounded Subscription Rates

Members are reminded that it is possible to compound for all future subscriptions by payment of a single sum. By doing this, the member benefits by paying a much smaller sum than would be required in separate annual fees, particularly if it ever becomes necessary to increase the annual subscription.

The Society benefits by having the immediate use of a lump sum; these payments are invested and credited to a special account in readiness for future use in purchasing headquarters for the Society or for similar purposes.

The present compounded subscription rates are as follows:

Age	Members		Fellows, Associate Fellows and Senior Members	
	£	$	£	$
20–25	41	115	50	140
26–30	37	105	45	126
31–35	34	95	40	112
36–40	31	87	36	100
41–45	28	79	33	92
46–50	26	73	30	84
51–55	23	64	26	73
56–60	21	59	24	67
over 60	18	51	21	59

It is hoped that as many members as possible will take advantage of this method of payment.

JOINT ACTIVITIES

European Symposium on Space Technology

It is now recognized by some, although the idea is as yet far from general, that a second Industrial Revolution is under way. The original Industrial Revolution, based on the common metals and fossil fuels, has brought much progress and prosperity and Europe was in the forefront of this progress.

Today, a new surge of technical progress is gathering momentum. This seems to be based on nuclear energy, applications of new metals, control engineering, and developments in computation techniques. Probably of still greater significance in this second Industrial Revolution is the development of space research and technology.

The new industrial era, destined to bring prosperity and power to those who embark upon it, is bound up in a rather special sense with developments in space technology which may indeed be said to play a catalytic role in the process. Any nation or group of nations which abstains from proper participation is likely to pay severe penalties.

Europe in its own interests *must* join the U.S. and the U.S.S.R. in this field. To this end the European Spaceflight Symposium is being convened, in the hope that the realization of the facts of the situation may become more general and a suitable response will be evoked to a challenging situation.

The symposium is being organized by the British Interplanetary Society with the co-operation of other principal European astronautical societies. It will take place in the Council Room of the Federation of British Industries, 21, Tothill Street, London, S.W.1, from Monday, **26 June** to Wednesday, **28 June, 1961**, inclusive.

Emphasis is being placed on the need to establish a spaceflight industry in the Western Europe/British Commonwealth area. It is hoped that directors and senior executives of leading aircraft, chemical, electronics, financial, metallurgical, and telecommunication firms will attend from the Continent, and so arrangements are being made for simultaneous translation of the proceedings.

The registration fee for attendance at the symposium is £5 5s.; this will also cover the cost of such preprints as are issued. Applications should be sent to The Secretary, British Interplanetary Society, 12, Bessborough Gardens, London, S.W.1.

Notes for Lecturers

It is proposed to have two distinct types of lecture, namely:—

A. Papers read in full by the author, followed by a short discussion. The first two days of the Symposium will be allocated to presentations of this particular type.

B. Papers whose full text will be distributed two days before the lecture. The author will only read a summary and this will be followed by a discussion lasting thirty or forty minutes. Those wishing to participate in the discussion will be invited to give their names before the lecture, to help the Session Chairman to steer the debate.

The themes or subject of papers will fall under two main headings—*Technical* and *European Collaboration*.

The first two days will be allocated to technical lectures presented in Form A defined above. The third day will be reserved for discussions of European Collaboration, the papers being presented in Form B.

The Technical group of papers may be sub-divided into two sections, namely:

(1) Future trends and development of spaceflight; review of future possibilities, related to West European capabilities, but taking into account advanced studies which need not call for very considerable budgets. Suitable subjects would be structural and propulsion problems and control engineering, treated from a practical technological angle, consistent with West European capabilities.

(2) Applications of space technology, particularly in the civil field. A good example would be orbital communication systems, but possibly on a scale less ambitious than some of the projects envisaged in U.S.A., *e.g.*, the stationary type of active satellite.

It is not intended that the Symposium should include papers concerned with some highly specialized aspect of astronautics or with a particular detail of vehicle or satellite design.

Papers under the heading of "European Collaboration" would be of a somewhat different nature, in fact, authors would be asked to provide a "framework" for the discussion. The subjects would be of a more general and industrial nature dealing with the scope and objectives of an eventual European programme and its organization on a supra-national scale.

Twelfth International Astronautical Congress

The Twelfth International Astronautical Congress, which is being organized by the American Rocket Society, will be held in New York early in October, 1961 (probably in the week 1–7 October).

The Congress will be along the same lines as the Eleventh Congress, held in Stockholm this August and attended by over 900 persons. An even greater attendance is anticipated in New York next year; in addition to the many Americans and Canadians who may be expected to attend a Congress held in their own continent, delegates from all the principal astronautical societies throughout the world will be present as usual. The Congress has a threefold purpose: in addition to the technical sessions there will be the meetings of the General Assembly and committees of the International Astronautical Federation, and there will be various social activities. At former congresses, the social activities have usually included receptions given by the host society and by the City in which the Congress is held or by the Government, a banquet, special trips for ladies, an excursion or river trip, works' visits, and film shows. There is much to interest those who do not wish to attend the lectures.

The 1961 Congress is being held later in the year than usual because the American Rocket Society feels that many of those attending will also wish to be present at its Annual Meeting, which is scheduled to take place on 8–14 October, 1961. A large exhibition devoted to rocketry, space technology and related industries will be associated with the A.R.S. Meeting, and may also be open during part, at least, of the Congress period.

In previous years both America and Russia have arranged for important astronautical experiments to take place immediately before or during the I.A.F. Congress, and participants have been able to discuss the achievements with those in charge of the programmes. It is early yet to speculate on next year's plans, but the possibilities include attempts at manned spaceflight by both America and Russia, launching trials of the giant Saturn booster, and tests of ion rockets in space. We also hope that by late 1961 a decision will have been reached concerning the establishment of a European spaceflight programme, so that this can be discussed at the Congress.

The Congress should be a memorable occasion, particularly as it will be the first to be held outside Europe. It is hoped that it will be possible for a large British contingent to attend (over fifty went from Britain to the Stockholm Congress this year).

The largest item in the cost of attending the Congress will be the fare; this would normally be in the region of £150. However, it is now possible to charter aircraft for the trip from London to New York at a cost varying according to the size of the aircraft and the percentage loading; £90 may be taken as the approximate fare for the round trip.

The Society hopes to charter an aircraft to attend the Congress and at least part of the A.R.S. meeting. To do so economically, it will be necessary to fill all the available seats and to ensure that a plane can be booked for the dates in question, it will be necessary to finalize the arrangements early in 1961. Any member who is thinking of attending the New York Congress is therefore requested to notify the B.I.S. Secretary as soon as possible, so that an estimate can be obtained of the number of persons who would take advantage of the charter flight.

It is obviously in everyone's interest that a full complement of travellers should be obtained. Under the regulations laid down by the U.S. Civil Aeronautics Bureau, participation in flights such as this is limited to members of recognized groups qualifying according to certain standards. It seems likely that it will therefore be necessary to limit participation strictly to members of the Society.

We must emphasize again that it is highly desirable that a preliminary list of those likely to take advantage of the charter should be drawn up as soon as possible, so that preliminary negotiations with the airline can be begun. It is not necessary at this stage for members (or others) to make a firm commitment or forward any money. All applications received by the Secretary will be numbered, and available seats in the aircraft will be allocated in the order in which the applications were received. Those applying will be notified at a later date of the exact fare and other arrangements. The possibility of payment of fares being by instalments or on deferred terms is being investigated.

It is suggested tht members on the Continent might also wish to consider the possibility of travelling to London and then taking advantage of this chartered flight.

Presentation of Papers

The American Rocket Society has not yet issued any regulations concerning the submission of papers, but these are likely to be along the lines of those for previous congresses. It is the intention of the Papers Committee to apply stricter standards in deciding whether contributions are acceptable or not.

All papers must be original contributions that have not been published in any form prior to the Congress. They may be written and presented in one of the official languages of the International Astronautical Federation (English, French, German and Russian), but it is recommended that English should be used if possible.

Prospective authors are invited to submit papers for presentation through the regional screening committees set up by the Member-Societies of the I.A.F. Each author is asked to notify the Chairman of the appropriate Committee of his intention to submit a paper and should accompany this notification with a provisional title, and an abstract if possible. Final dates for submission of abstracts and manuscripts will be announced later, but it would be appreciated if those members of the Society who are considering writing a paper would immediately notify the Chairman of the I.A.F. Committee (Dr. L. R. Shepherd, British Interplanetary Society, 12, Bessborough Gardens, London, S.W.1) of the proposed title. If possible, a short synopsis should also be enclosed.

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